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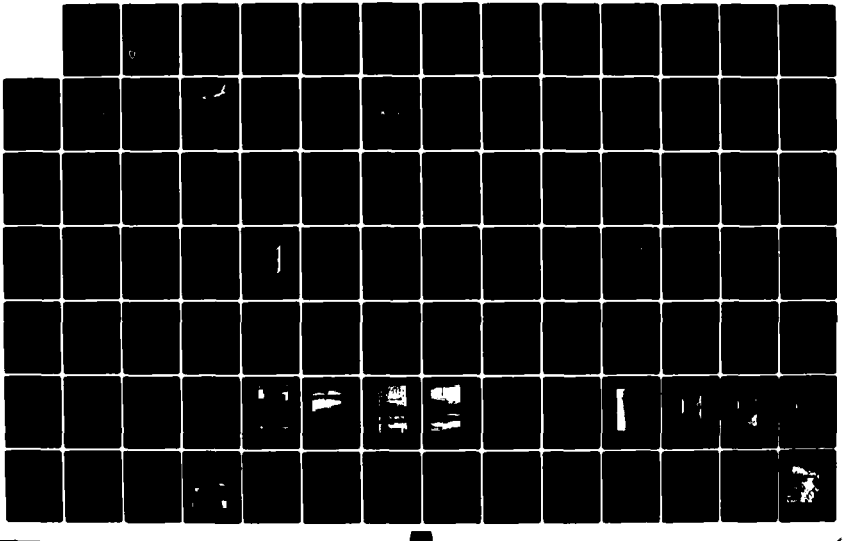

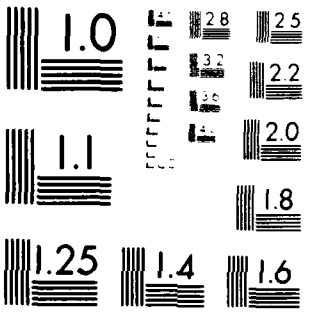

AVRADCOM
Report No. TR-82-F-1

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MANUFACTURING METHODS AND TECHNOLOGY (MANTECH) PROGRAM MANUFACTURING TECHNIQUES FOR A COMPOSITE TAIL SECTION FOR THE ADVANCED ATTACK HELICOPTER

J.V. ALEXANDER
R.E. HEAD

October 1981

FINAL REPORT

Contract No. DAAK50-78-G-0004
DO 0002

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UNITED STATES ARMY
AVIATION RESEARCH AND DEVELOPMENT COMMAND

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Ballistic Survivability			

20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

This manufacturing methods and technology program involved the study of estaolishing manufacturing methods for a Composite Tail Section (CTS) for the AH-64A Advanced Attack Helicopter by the wet filament wound, cocure process. The CTS was meant to replace equivalent metal components on the AH-64A while providing potential weight and cost savings. Ballistic tolerance against 23mm HEI-T was demonstrated, and VHF antenna

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20. performance was shown to be unaffected by the change to composite construction. The program was terminated for fiscal reasons just when the CTS design was completed and the tooling design was within one month of completion. This CTS concept offers significant benefits to the AH-64A helicopter program and should be reinstated at the earliest opportunity.

PREFACE

This report was prepared by Hughes Helicopters, Inc. (HHI) under U. S. Army Basic Ordering Agreement DAAK50-78-G-0004, Delivery Order 0002. The contract was sponsored by the U. S. Army Aviation Research and Development Command (AVRADCOM) and administered under the technical direction of Mr. James Tutka, AVRADCOM, with assistance from Mr. Nicholas Calapodas, Applied Technology Laboratory, Ft. Eustis, VA.

The technical tesks were conducted under the direction of HHI's program managers, Messrs. Nicholas Mocerino and Robert Head, and Mr. Sherwood Twitchell, the project engineer at Fiber Science, Inc., the major sub-contractor for tooling and fabrication.

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INTRODUCTION

In August, 1979, Hughes Helicopters, Inc. (HHI) was awarded a contract under a Manufacturing Methods and Technology (MM&T) program to develop and refine the manufacturing processes necessary to fabricate and test a set number of Composite Tail Sections (CTS) for the AH-64A Advanced Attack Helicopter (AAH). Anticipated benefits resulting from this MM&T program were reduced weight and cost, and improved survivability, reliability, and maintainability.

The CTS program as initially defined required that the aft portion of the tailboom, the vertical tail spar, and the horizontal tail be the primary composite structure. See Figure 1. Existing fairings from the prototype design such as those for the leading and trailing edges of the vertical tail, driveshaft cover, and aft tailboom fairing were to be transferred to the CTS. The CTS was to accommodate all existing mounts for the gearboxes, controls, tail wheel, etc.

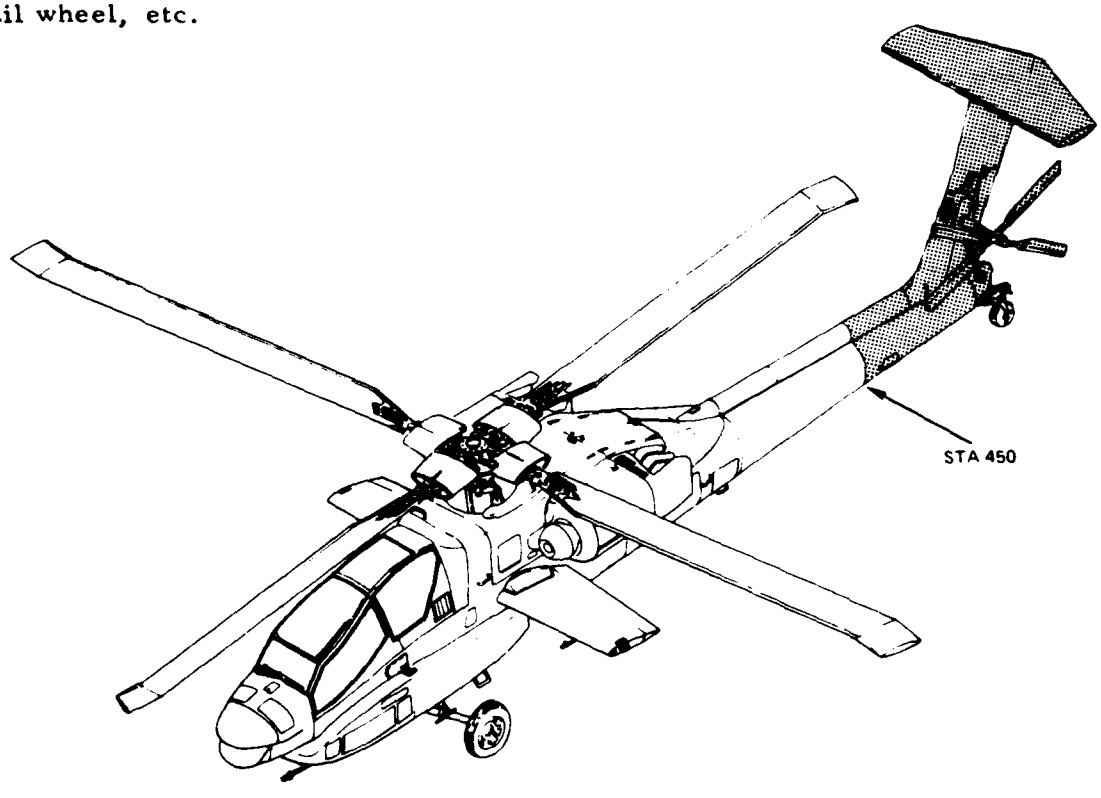


Figure 1. AH-64A - Initial CTS Configuration

The CTS technology was based on work HHI had done previously in concert with Fiber Science, Inc. (FSI) for a composite tail assembly for the AH-1G helicopter (Reference 1), a composite main rotor blade for the AH-1G (Reference 2), and a composite landing gear for the AH-64A (Reference 3). These components were fabricated by the wet filament winding, cocure process and this process was to be carried over into the CTS program. HHI had overall program responsibility, did the design refinement work, and was to conduct all laboratory and flight testing. FSI was responsible for optimizing the tooling and fabrication processes for the CTS components.

When the CTS program was proposed, the prototype metal tailboom for the AH-64A had a fold joint at Fuselage Station (Sta) 450 to aid in an air transportability in C-141 cargo planes. However, the Air Force lengthened the fuselages of all its C-141s making the AH-64A tail folding no longer needed. In the interest of structural efficiency, the fold joint was replaced by a tension bolt attachment. The prototype tail assembly was a T-tail configuration with the horizontal stabilizer rigidly attached to the top of the vertical tail.

This was to be the configuration for the composite tail assembly -- a geometric copy of the prototype metal tail section.

AH-64A flight tests that were made just before the CTS design work began showed that the helicopter's flying qualities could be markedly improved by changing from the T-tail configuration to one having a controllable stabilator mounted at the base of the vertical tail. When this change was made, the tail rotor was moved up the vertical tail to have clearance from the new position of the stabilator. The composite tail design matched this flight test configuration from the beginning (see Figure 2).

As the design work progressed, it became apparent that important benefits could be attained if the composite portion of the tailboom was extended further forward. A contract modification was negotiated to extend the composite structure 80 inches forward and make the metal-to-composite joint at Sta 370. This was the farthest forward point the composite structure could go without compromising the existing design of equipment bays, access doors, etc. It was determined that a shear bolt attachment at this point would be a production manufacturing break.

Other modifications were made to the CTS design to accommodate the findings of the prototype flight test program and the composite flexbeam tail rotor design program that were going along in parallel. These included softening the torsional stiffness of the tailboom to avoid a resonance with the main rotor 4-per-rev frequency, increasing the torsional strength of the tailboom and the sideward bending strength of the vertical tail to support

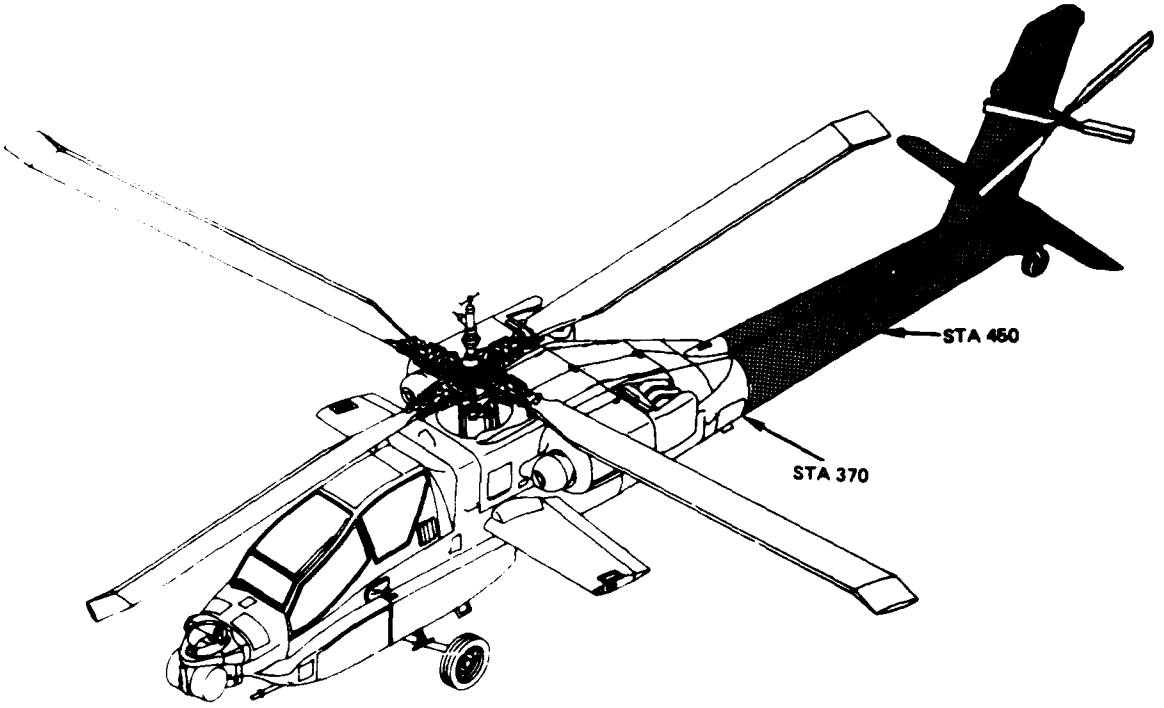


Figure 2. AH-64A Final CTS Configuration

the composite tail rotor that had 16 percent more thrust capability than the flight test metal tail rotor, and changing the vertical tail incidence and camber to unload the tail rotor in cruise flight to reduce its flapping motion for better fatigue life.

So what started out to be essentially a one-for-one replacement of metal by composites turned into a sophisticated CTS design that improved dynamics, permitted the use of the larger tail rotor needed for high altitude controllability when the T700-GE-701 engine is installed, and made a spectacular reduction in parts count while reducing the weight and cost. In this CTS configuration, the gearbox, control system, and tail wheel mount points were held inviolate, and the drive shaft covers and vertical tail leading edge fairing were retained. A new trailing edge fairing for the vertical tail spar was required by the modified incidence and camber.

A tradeoff study was made to determine the best design concepts for the three CTS components: tailboom, vertical tail, and stabilator. For the tailboom and vertical tail spar, a monocoque honeycomb sandwich shell with

frames located at high load points was determined to be optimum, consistent with good design practice and the wet filament winding manufacturing process. The skins of these components were selected to be hybrid graphite/Kevlar/epoxy. The best stabilator configuration proved to be a skin/spar/rib arrangement with graphite/epoxy used for the spars and central ribs, and Kevlar/epoxy for the skin and for the nose and tail portions of the ribs.

In parallel with this design refinement work, FSI established refined processes for fabricating the three CTS components, predominantly by the wet filament winding, cocure process that had been used successfully in a wide variety of previous programs. This work included analytical and experimental assessment of mandrel shapes to accommodate the specified winding angles, frame and hardpoint inserts, etc. Based on these activities, tooling suitable for building 50 shipsets of the components was designed.

At the time that the CTS program was terminated, the design was complete. The tooling design was in the final stages, and tool fabrication was ready to start. Material properties tests in support of design had been completed. Antenna pattern tests had been conducted to show that the composite structure had no appreciable influence on the radiation characteristics. Ballistic damage tests, conducted at the Army's Ballistic Research Laboratory, had shown that simulated composite tailboom components were survivable against 23mm HEI-T projectiles equipped with MG-25 fuzes. Lightning survivability tests had demonstrated the suitability of the design. Both the ballistic and lightning tests were sponsored by HHI.

The analysis of the CTS showed that in comparison with the metal* components it would replace, there would be:

- 71 pounds weight saving
- $41,000 design to unit production cost saving (1981 dollars)
- 269 parts count saving
- 9047 fastener count saving
- 55 feet per minute vertical rate of climb increase from weight reduction
- $1.1 million fuel saving over life of the fleet

*AH-64A weight and cost values for metal components are those quoted for the second quarter of 1981.

DESIGN REQUIREMENTS

The Composite Tail Section (CTS) consists of three major components:

- Tailboom
- Stabilator
- Vertical Tail

plus fairings, covers, steps, handholds, lights, and antennas.

The basic geometry of the CTS is shown in the lines drawing, Figure 3.

The strategy for optimizing the design of the CTS to match optimized techniques used in building it is to create a low cost, light-weight configuration while providing:

- External Geometric Similarity
- Accommodation of Existing Gearboxes, Controls and Tailwheel
- Maximum Incorporation of Filament-Reinforced Plastic Structure
- Aerodynamic Compatibility
- 4500-Hour Service Life
- Ballistic Survivability
- Lightning Survivability
- Reduced Parts and Fastener Count
- Interchangeability at Fuselage Station 370
- Accommodation of Existing Antennas, Lights, Mission Equipment Kits

In addition, the configuration must accept either the 112-inch diameter composite flexbeam tail rotor or the 110-inch diameter prototype metal tail rotor.

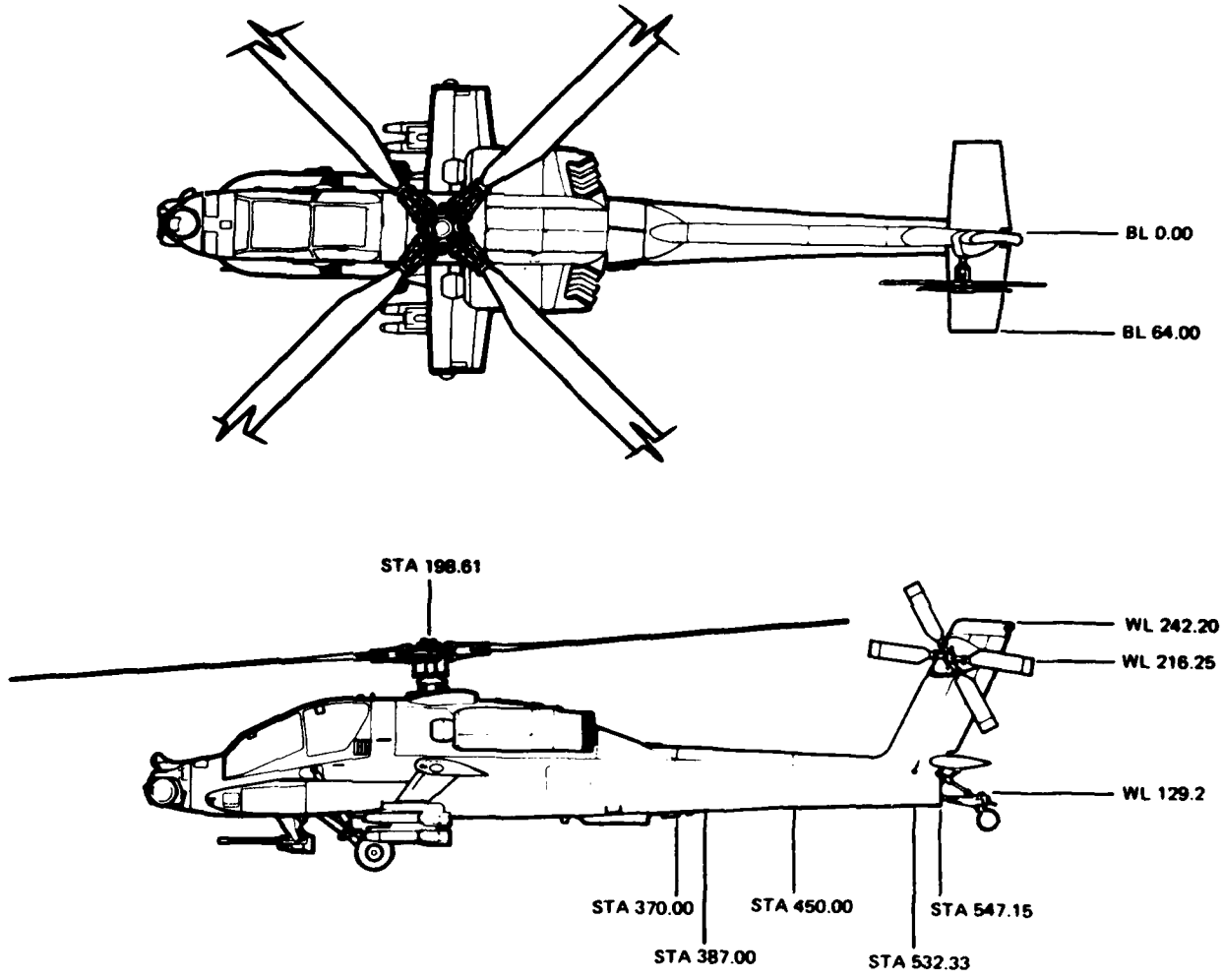


Figure 3. AH-64A Lines Drawing

The basic AH-64A description and all its design criteria remain the same as those described in Reference 4, the Design Criteria Report for AH-64A vehicle, with the exception of minor geometry changes to the stabilator and vertical tail (see Design Refinement section).

For reference, the flight envelope for weights at or below the Basic Structural Design Gross Weight (BSDGW) is presented in Figure 4. For weights above the BSDGW, the maximum vertical load factor is reduced by the ratio of the BSDGW divided by the gross weight for the particular condition up to the maximum alternate gross weight. In no case is the load factor less than 2.0. The sideslip envelopes for the AH-64A are presented in Figure 5 for the static limit, transient limit, and emergency limit.

The requirements for CTS stiffness to assure good dynamic properties are given in Figures 6, 7, and 8. These are chosen to place the stabilator roll and yaw assymetric natural frequencies at 30 Hz while avoiding the main rotor 4P, main rotor 8P, and tail rotor 1P excitation frequencies that occur at 19.3, 38.6, and 23.5 Hz, respectively.

The basic external loads for the composite tail section are used for static and failsafe analysis and, where specified, for weighted fatigue. Oscillatory load levels from the flight strain survey weighted fatigue level of ±10 percent of limit load for the fuselage and ±30 percent for the vertical stabilizer and stabilator are used. These loads are the same as those specified for the basic AH-64A and are calculated for the flight maneuver conditions specified in Reference 5, and in Army-approved deviations to Reference 5.

One of the driving forces behind the increased use of composites is the increase in reliability; i.e., the decrease in failure rate. Structural failures can be classified according to severity and cause.

Severity of failures can be classified as minor, major, and critical. Minor failures merely require maintenance, and can generally be termed discrepancies. Major failures represent a threat to personnel and equipment and may result in a degradation of performance. Critical failures may result in loss of control of the aircraft.

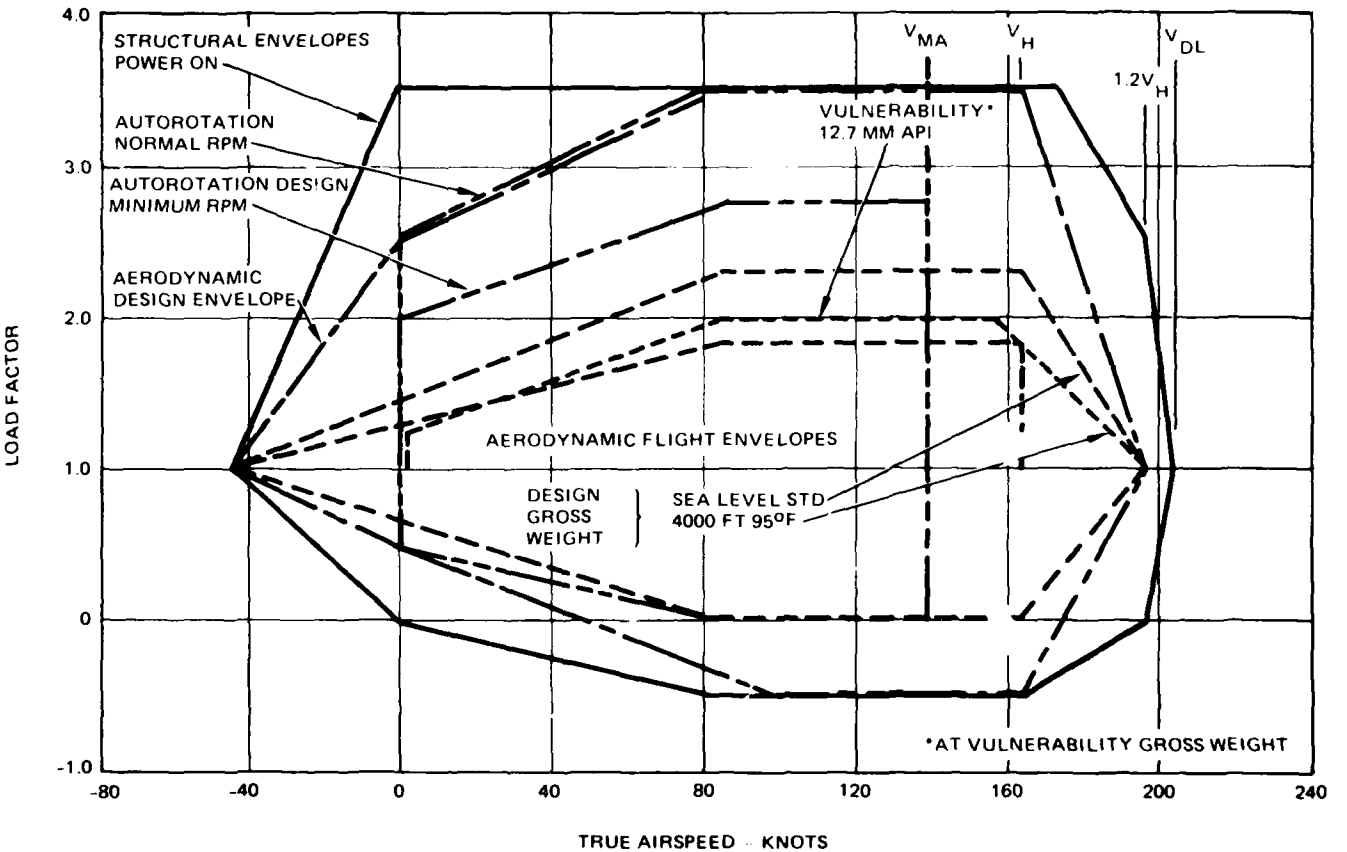


Figure 4. Flight Load Factors Versus Airspeed (Basic Structural Design Gross Weight Except Where Noted)

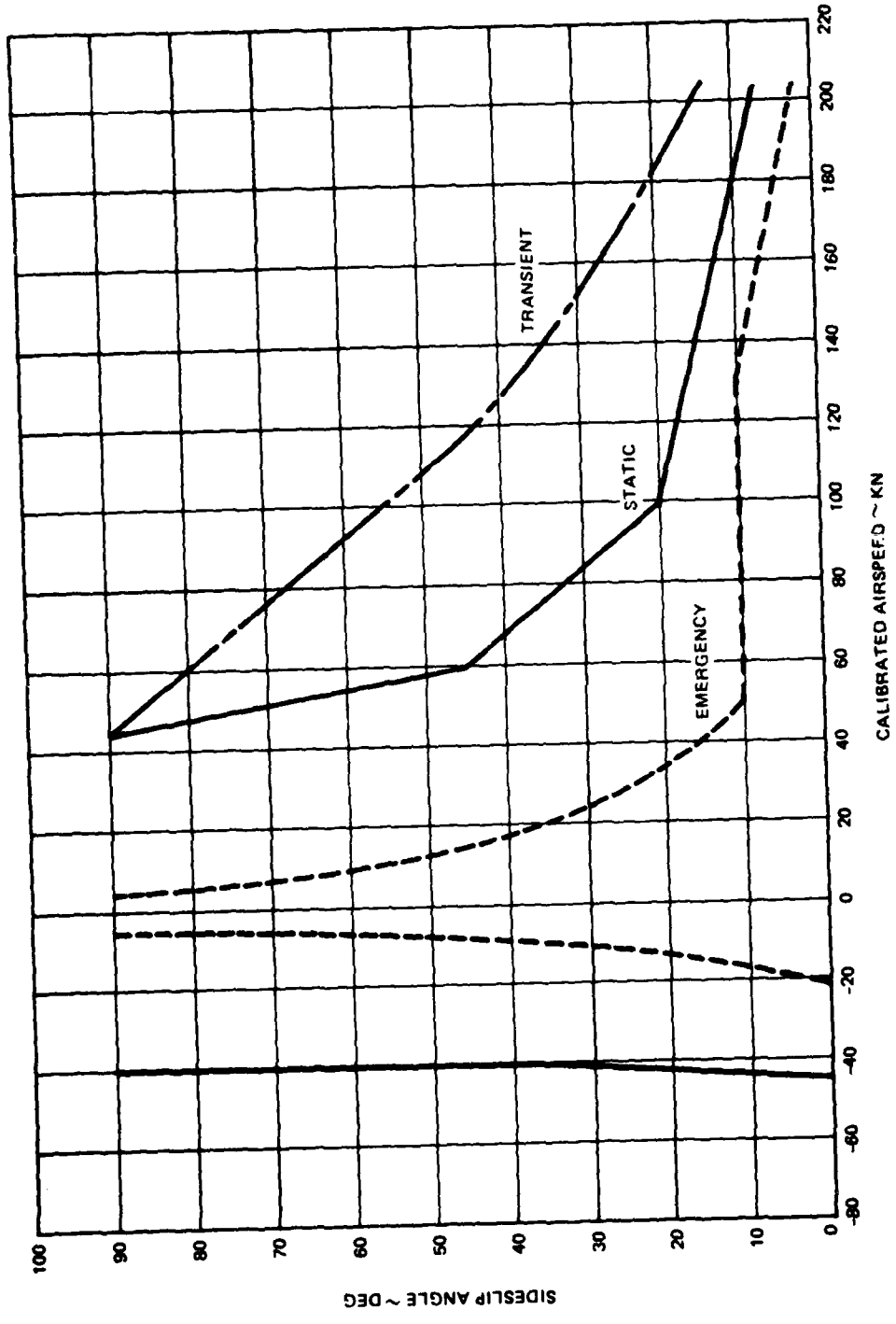


Figure 5. Sideslip Envelopes

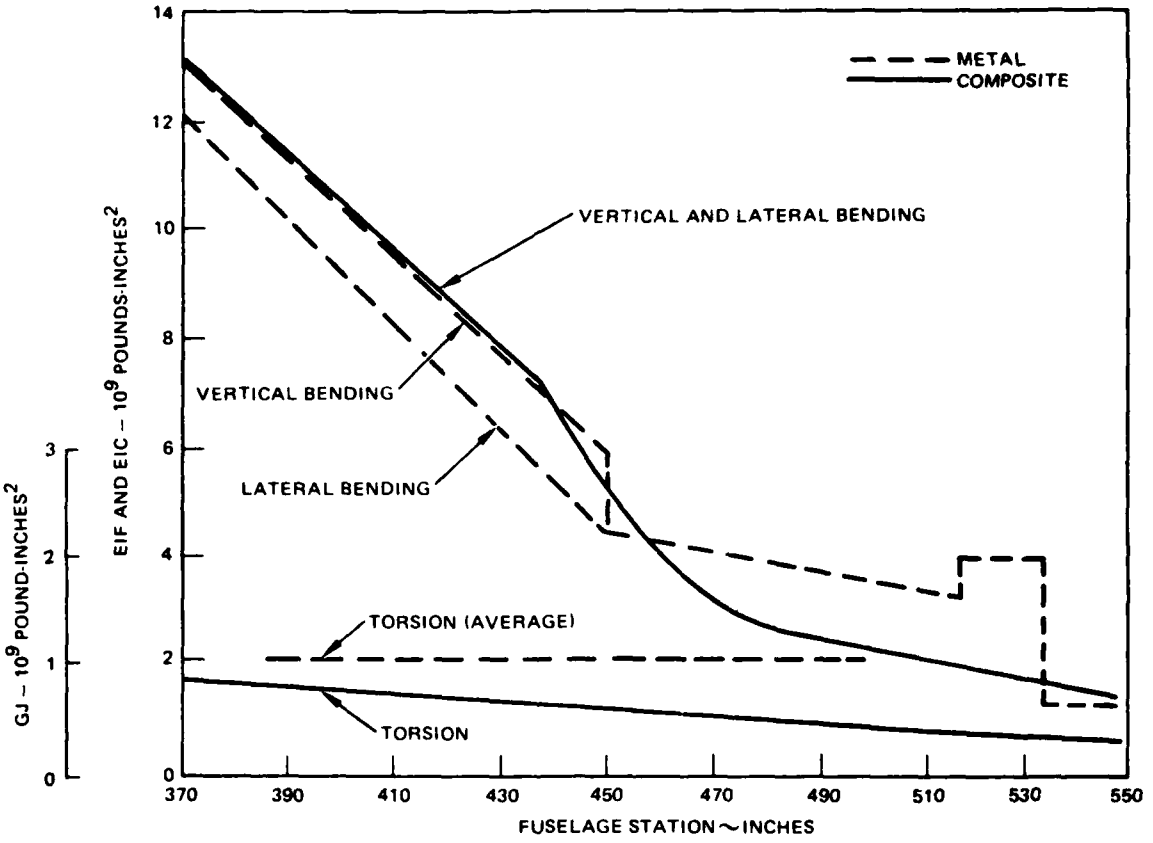


Figure 6. Tailboom Stiffness

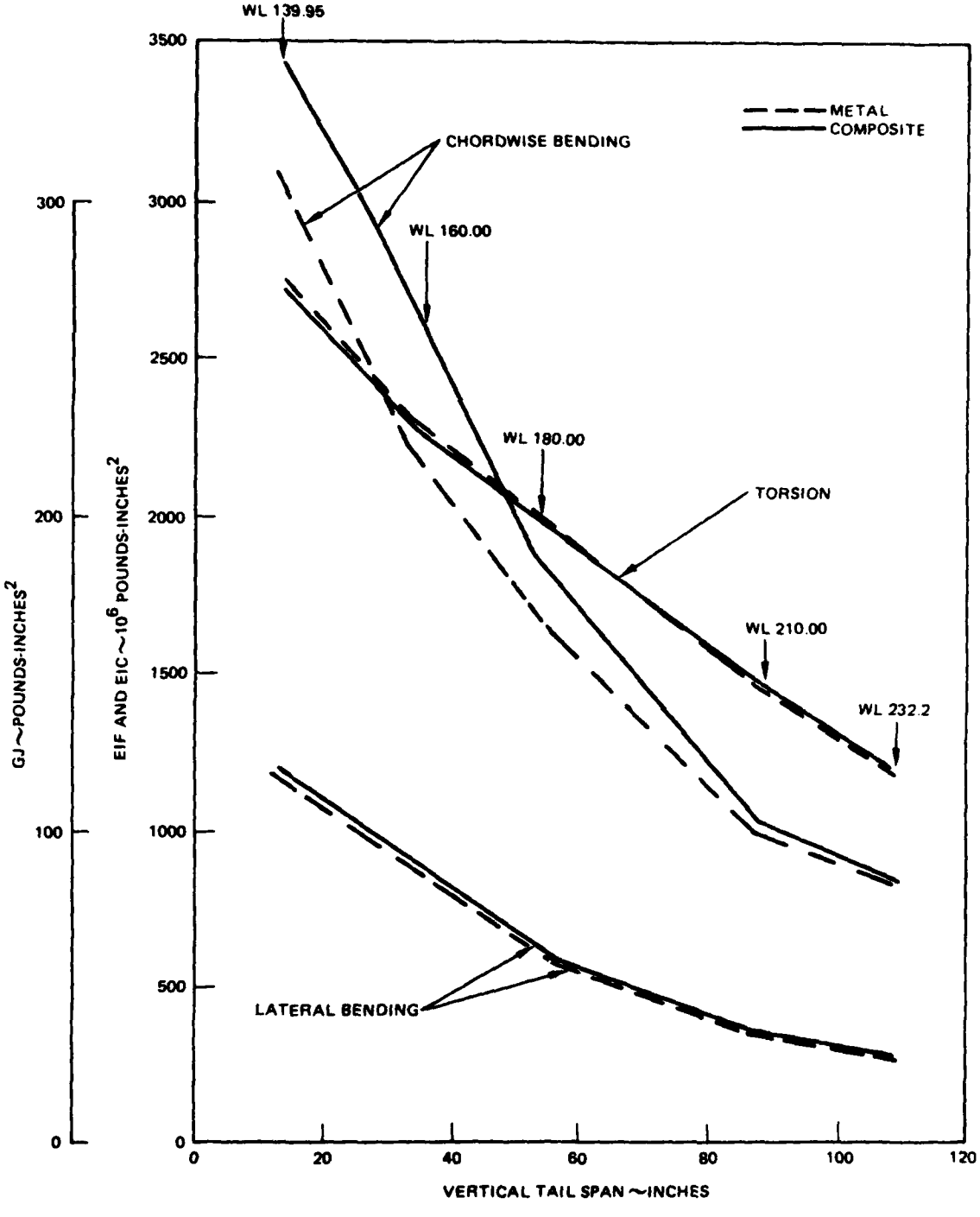


Figure 7. Vertical Tail Stiffness

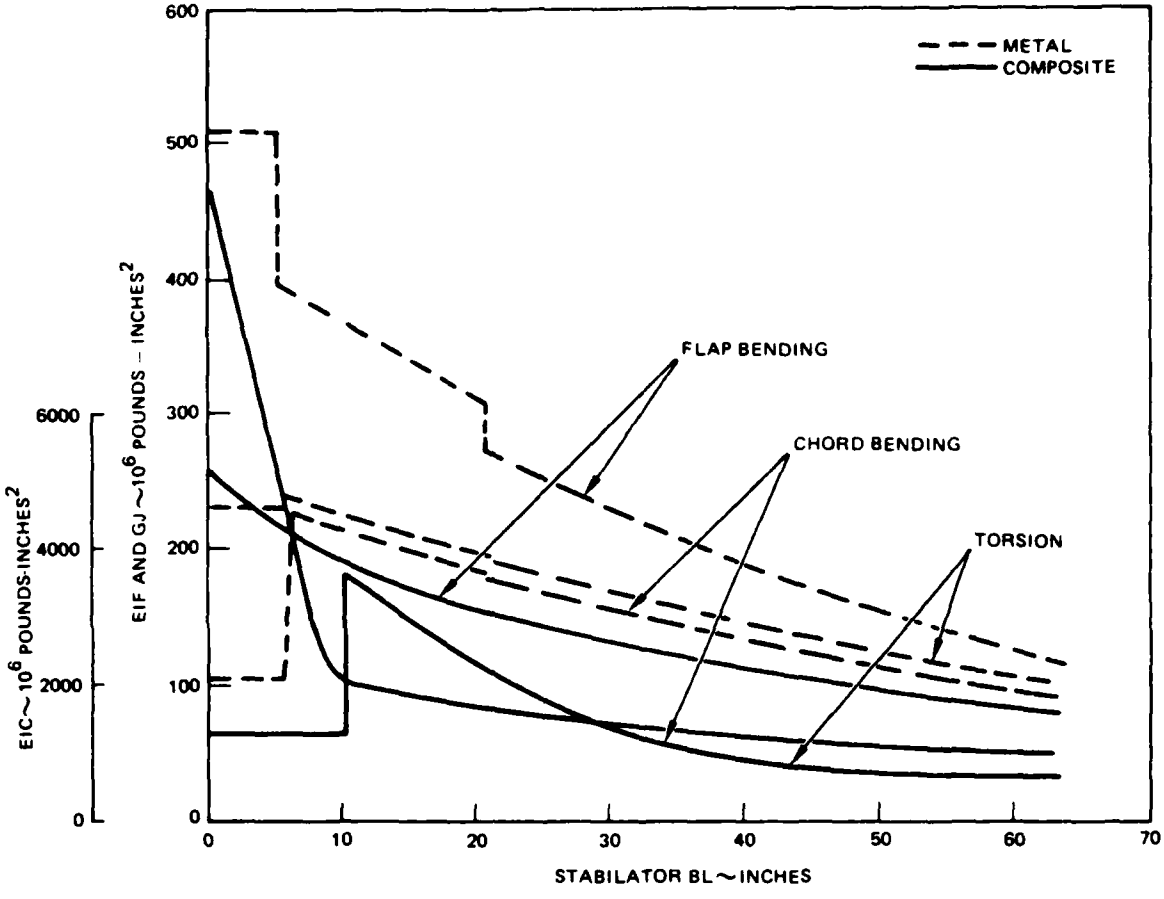


Figure 8. Stabilator Stiffness

Classification by cause divides failure into inherent or induced modes. Induced failures generally include any damage caused by external influences such as foreign objects, by certain environmental factors, or by conditions outside of the design envelope (such as overload conditions). Inherent failures include anything that is not induced such as fretting, cracking, etc., that occur at load levels within the design envelope. Examples of induced and inherent failures are given in Table 1.

TABLE 1. EXAMPLES OF STRUCTURAL FAILURES

Inherent Failures

Minor

- Fretting
- Delaminations
- Change in bolt preload

Major

- Formation of small crack
- Loss of primary fastener

Critical

- Loss of multiple primary fastener
- Growth of large crack

Induced Failures

Minor

- Scratch or gouge
- Delamination
- Lightning damage

Major

- Deep gouge
- Small ballistic perforation

Critical

- Large Ballistic perforations

DESIGN REFINEMENT

The AH-64A Composite Tail Section (CTS) is designed according to the existing metal tail section design criteria. Based on the success that HHI had experienced with wet filament winding in other aircraft primary structure programs for the Army, this process was selected for fabricating all the major components.

Design refinements include optimization of weight, mass distribution, inertial properties, stiffness distributions, natural frequencies, vibratory loads, fuselage vibrations, ballistic survivability/tolerance, anticipated service life, maintainability/reliability, acquisition cost, and life cycle cost. The design is in accordance with the AH-64A system specification AMC-SS-AAH-H10000A and meets the following restraints:

- The aluminum fuselage/CTS interface is a permanent joint at Fuselage Station (Sta 370).
- CTS stiffness distributions are dynamically compatible with the test aircraft.
- The CTS interfaces with existing operational hardware of the AH-64A (i.e., gearboxes, tail rotor drive shaft, control components, landing gear, etc.).
- The external geometry of the aluminum baseline configuration is maintained.
- The antennas are functionally integrated within the tailboom and vertical tail.

This section describes the design features of the CTS.

The CTS that consists of three major components -- tailboom, stabilator, and vertical tail -- mates with the AH-64A airframe in a manufacturing splice at Sta 370. Figure 9 is an isometric sketch that shows the configuration of the CTS and the associated fairings, covers, and external hardware that mate with it. Assembly and major subassembly drawings of these components may be found at the back of this report (Figures 95 through 99).

Before this configuration that is based primarily on sandwich wall monocoque construction was selected, a predesign study investigated other structural concepts. The primary contender was a skin, stringer, frame design of

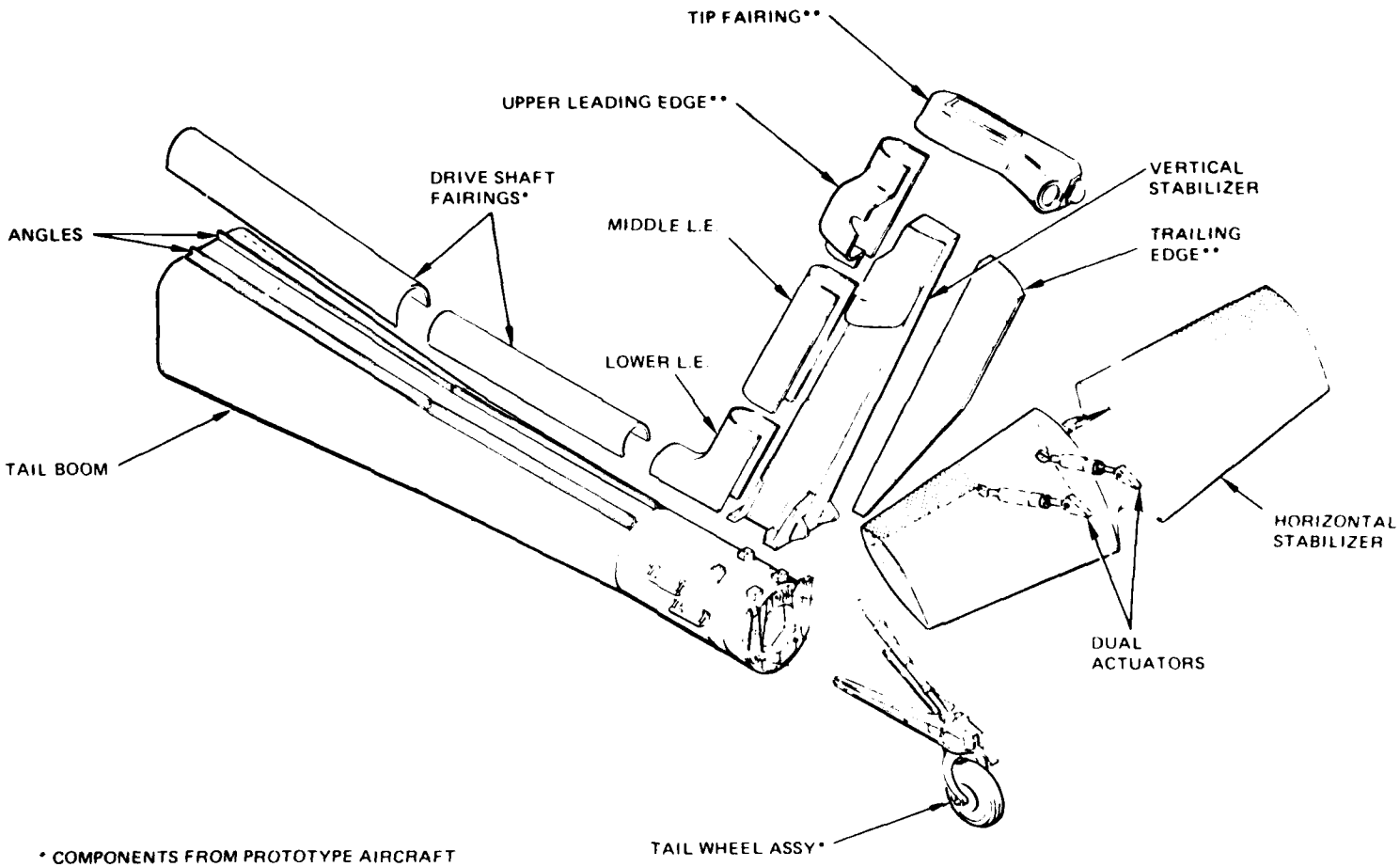


Figure 9. Composite Tail Section with Adjacent Fairings, Closeouts, and Internal Hardware

the tailboom and vertical tail spar. Structurally, this alternate design was equally acceptable, but the sandwich concept was selected as being more amenable to the wet filament winding manufacturing process.

TAILBOOM

The composite tailboom shown in Figure 10 retains most of the baseline metal fuselage configuration. It transitions from a flat-topped ovaloid at Sta 370 to a circular cone from Sta 450 to Sta 530. From Sta 530 aft, the composite tailboom maintains a constant diameter, differing from the continuing cone shape of the baseline configuration.

The composite tailboom is composed of the following major components (see Figures 95 and 96:

- Structural Shell
- Station 387 Frame
- Forward Fitting Assembly
- Intercostal, Upper
- Intercostal, Lower (2)
- Intercostal, Jack Fitting
- Rear Frame

Other details that are attached onto this assembly are:

- Access Steps, Upper and Lower
- Angle Fairing, Right and Left
- Jack Point Fitting

Hard points (local reinforcement and/or core filled honeycomb) are provided for the following details that are common to both CTS and metal baseline:

- Drive Shaft Fairing Hinges
- Drive Shaft Bearing Support Bracket

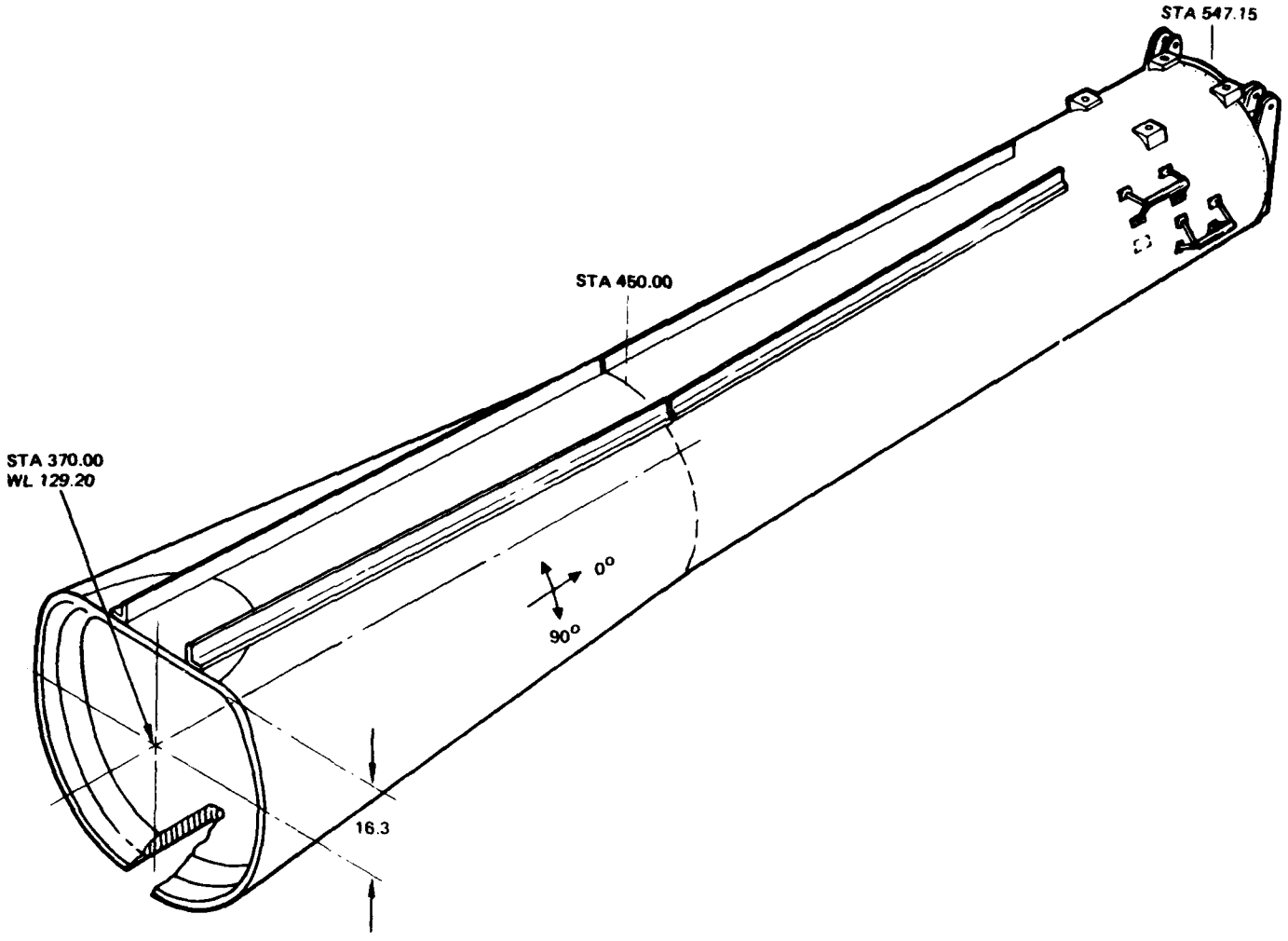


Figure 10. Tail Boom Configuration

- Drive Shaft Anti-Flail and Damper Support
- Flare Dispenser

The structural shell of the tailboom is a graphite-Kevlar/epoxy sandwich with balanced helically wound skins and 0.50 inch thick Nomex core. The inner and outer skins converge at the front end of the tailboom (Sta 370) and at the aft, constant diameter section to form solid laminates as Figure 11 shows.

The base sandwich skin thickness is 0.043 inch and the winding angle is 8° ±1° at Sta 370. Skin thickness and wrap angle increase from the forward to aft end of the tailboom. Hoop windings (90°) stabilize the spiral windings. The skin is wound from a 50-50 ratio of graphite/Kevlar rovings. This winding pattern is tailored to provide the required bending and torsional stiffness and strength.

The solid laminate attach flange at Sta 370 is interwound with ±45° graphite fabric and 90° E-glass to a total flange thickness of 0.172 inches. The interwound plies, in addition to providing the necessary bearing area, are structurally balanced in thickness and elastic modulus to evenly distribute load into the two skins. Figure 12 shows how this solid laminate region is attached to the forward airframe by splice plates and a double row of HiLok fasteners.

The solid laminate at the aft end of the tailboom (sta 530, aft) is also made up of interwound fabric plies tailored for bearing strength and balanced load distribution.

Local attach points and cut-outs employ additional interwound plies of fabric to provide load-carrying structure in these high stress areas. The honeycomb core in these areas is filled with syntactic foam to stabilize the core, anchor the fasteners, or provide compressive strength to support fastener loads.

The materials used to fabricate the tailboom are:

- Graphite Roving: T-300, 3000 Filament Count
- Kevlar Roving: Kevlar-49, 1420 Denier
- Graphite Fabric: T-300, 3000 Filament count, Eight Harness, Satin Weave
- E-glass Fabric: 1581 Glass Fabric

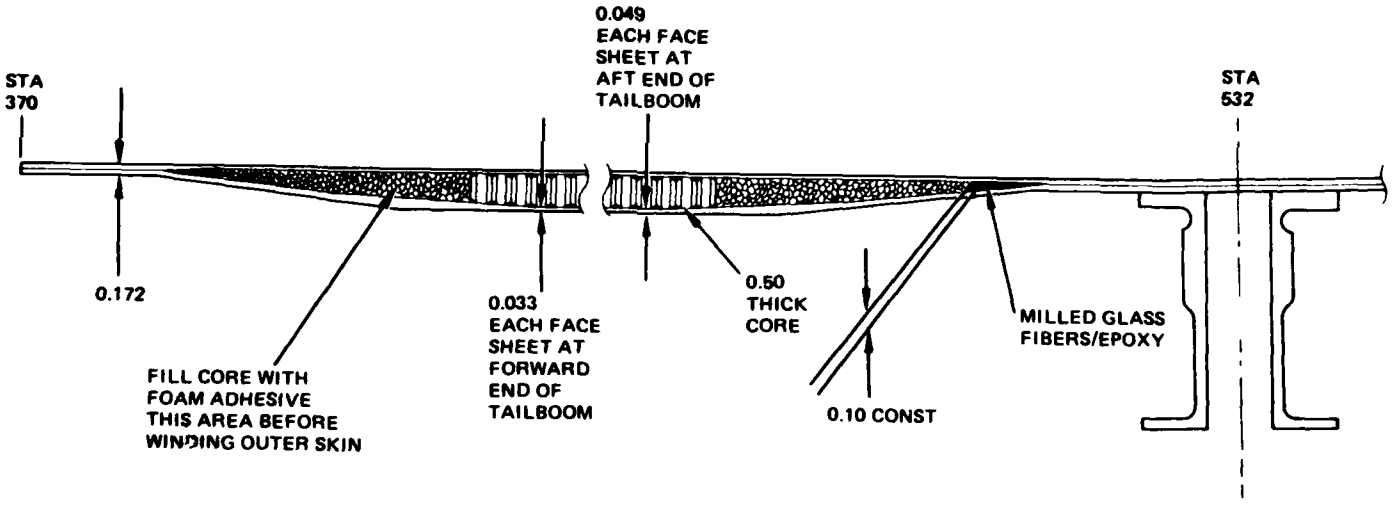


Figure 11. Tail Boom - Structure Details

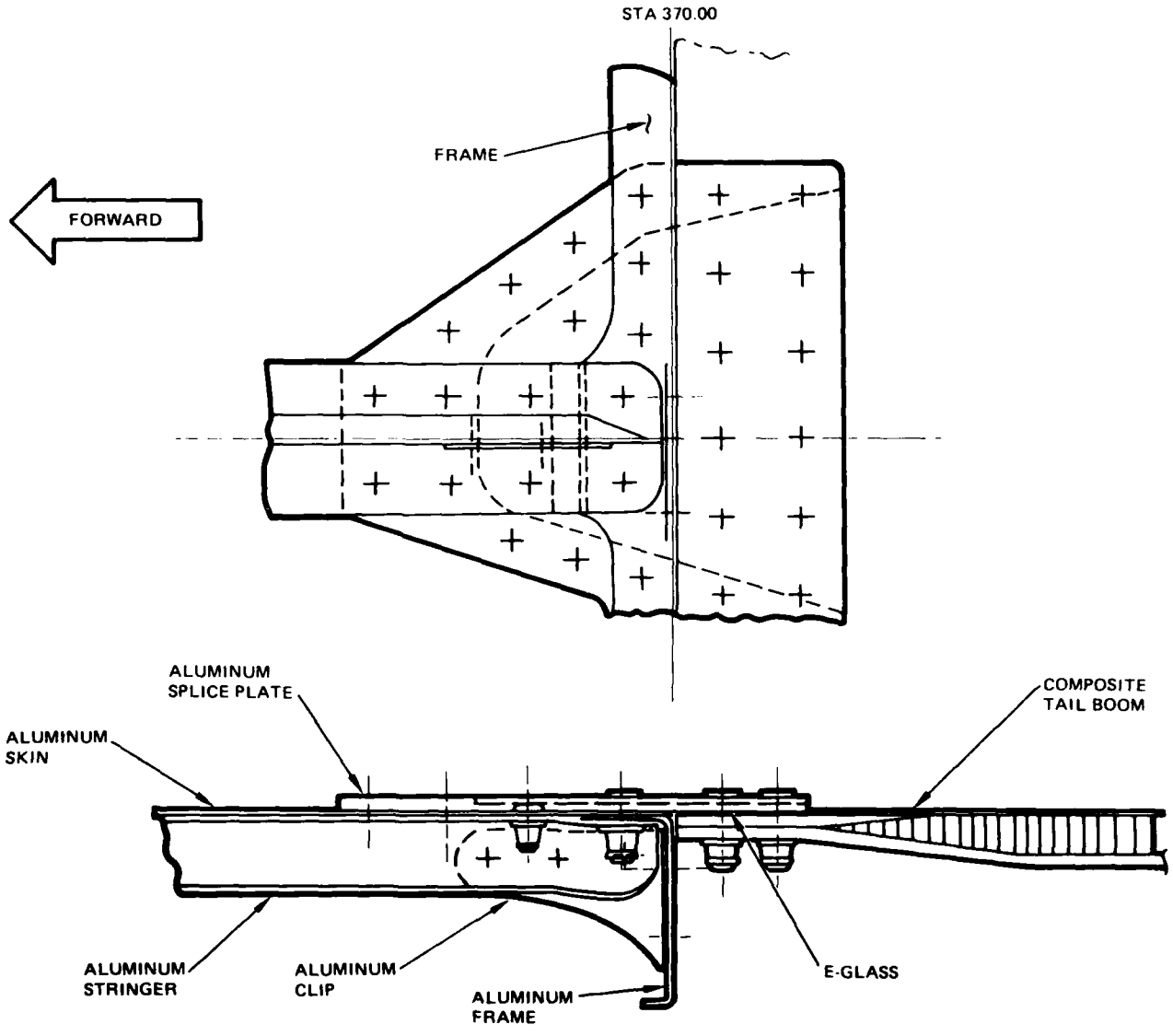


Figure 12. Metal/Composite Manufacturing Splice, Sta 370

- Epoxy Resin: APCO 2434/2347
- Core Adhesive: FM 123-5 Film Adhesive
- Core: Nomex HRH - 10/0X - 3/16

There are three graphite/epoxy frames that are molded separately and then assembled into the tailboom. The forward one at Sta 387 is provided to stiffen the tail rotor shaft bearing support at this location. It is installed by a secondary bonding process. The two back-to-back frames centered at Sta 532.33 are made into a subassembly unit consisting of these two frames, a connecting strap bonded around them, and four graphite/epoxy intercostals that distribute tail wheel loads into the tailboom. This subassembly unit is wound into the tailboom shell structure to become an integral part of it. The tailboom in the region of this subassembly unit is shown in Figure 13.

A machined aluminum fitting is bolted between the two back-to-back composite frames after the tailboom is cured (see Figure 14). This is the fitting to which the forward bolts for the vertical tail attach.

The materials for the frames are:

- Fitting: 7075-T6 Aluminum Alloy
- Frames: Graphite Fabric: T-300, 3000 Filament Count, Eight Satin Weave

  Glass Fabric: 1581 Glass Cloth

  Resin: APCO 2434/2347

Four graphite/epoxy intercostals are located between the forward/frame assembly and the rear frame, (see Figure 13). Three of these intercostals transfer most of the tail wheel shock and strut loads into the skin. The balance of the load goes into the forward frame assembly through a row of shear bolts. The fourth intercostal transfers jacking and tie-down loads into the tailboom. The materials for the intercostals are :

- Graphite Fabric: T-300, 3000 Filament Count, Eight Harness, Satin Weave
- Epoxy Resin: APCO 2434/2347

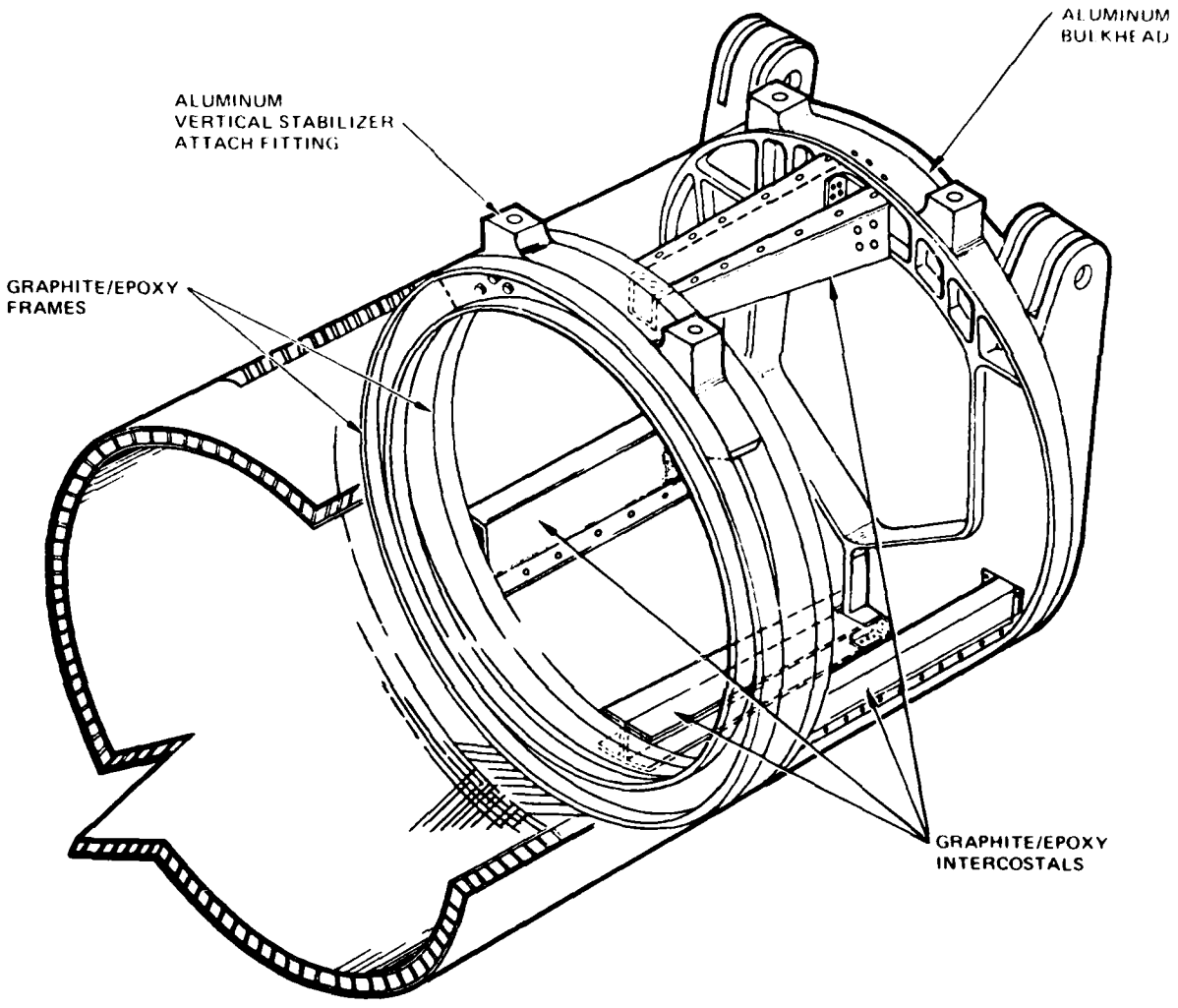


Figure 13. Tail Boom

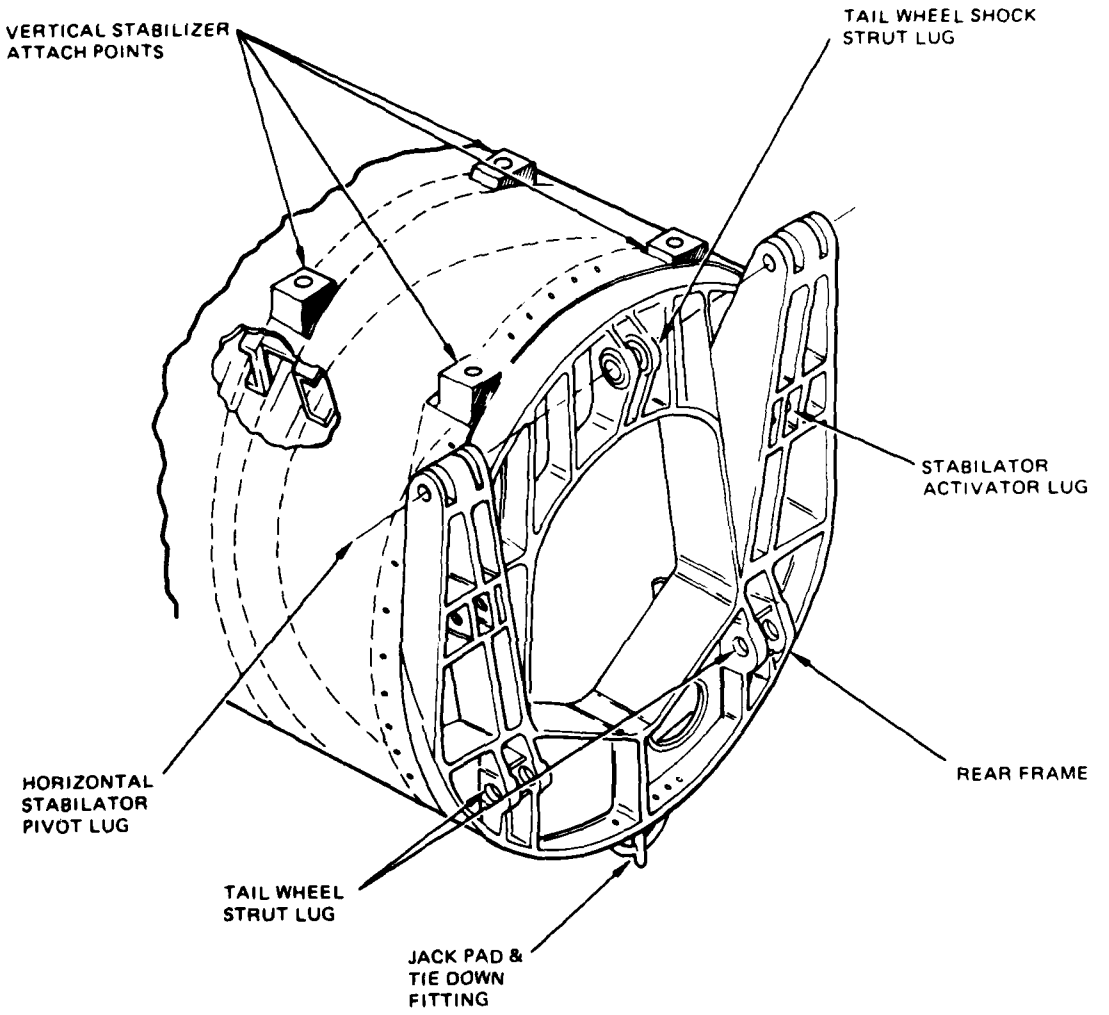


Figure 14. Aft End of Tailboom

The rear frame, Figure 14, is a machined aluminum fitting that has multiple functions. It contains the tail wheel assembly attach points, stabilator pivot points, and the stabilator actuator attach points. It provides the two aft points for vertical tail attachment and anchors the aft jacking fitting. The frame, made from 7075-T6 aluminum alloy, is bolted into the tailboom with HiLok fasteners.

VERTICAL TAIL

The geometry of the composite vertical tail is:

Area (from boom ℄) - ft^2	32.2
Span (from boom ℄) - in.	113.0
Tip chord - in.	35.1
Chord of section at W.L. 196.0 - in.	43.5
Root chord (at boom ℄) - in.	44.0
Geometric aspect ration (based on span from boom ℄)	2.5
Airfoil	NACA 4415 modified
Leading edge sweep - deg	29.4
Rudder - (mean) percent chord	55.0
Rudder deflection - deg	4.0
Rudder tab - (mean) percent chord	25.0
Rudder tab deflection (from rudder chord line) - deg	18.0
Geometric incidence - deg	0.0
Aerodynamic incidence - deg	9.4

The major structural element in the vertical tail is the spar box shown in Figure 15. Leading and trailing edge fairings and tip cap as shown in Figure 9 complete the assembly. The vertical tail's planform is unchanged

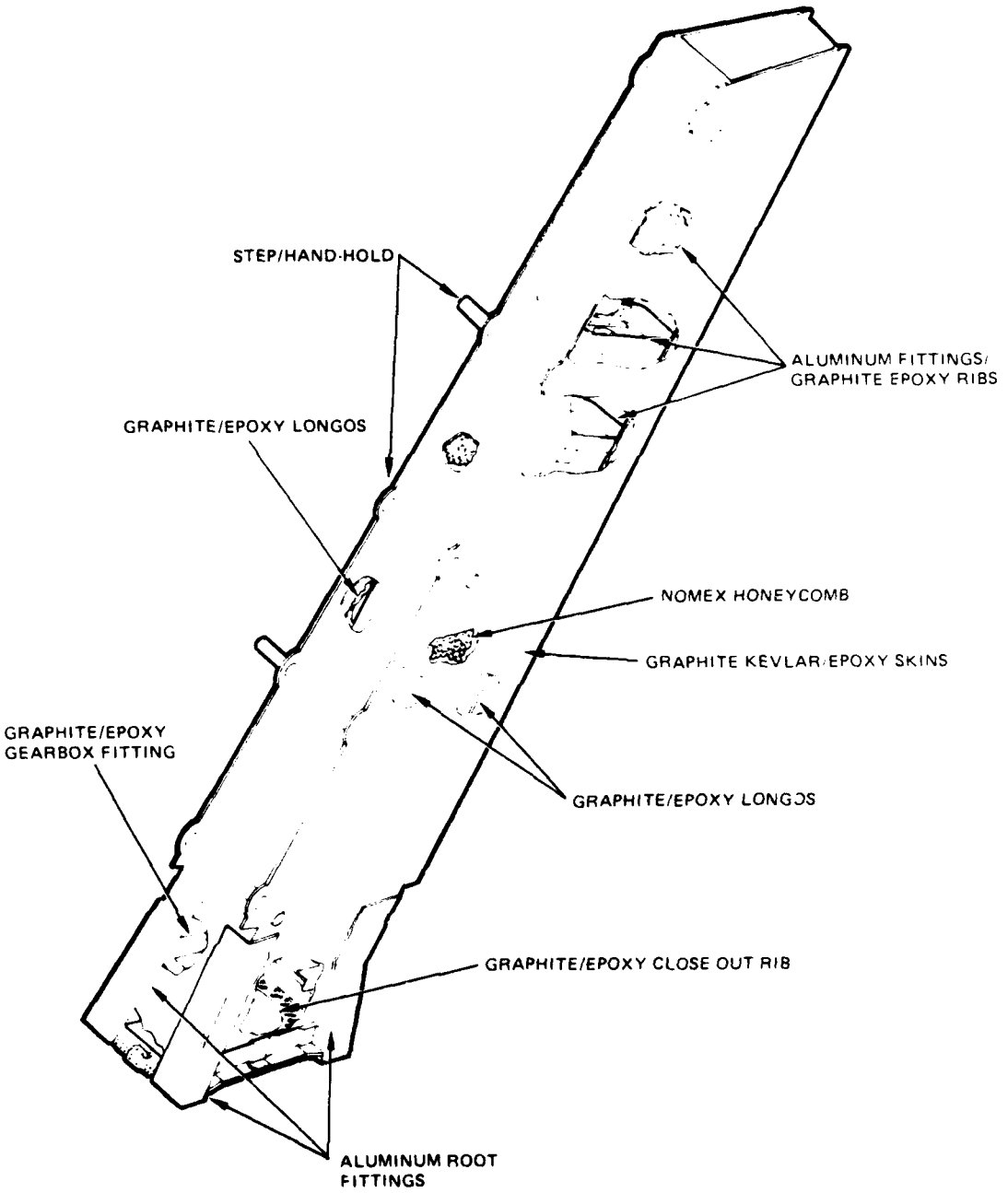


Figure 15. Vertical Tail Spar Box Structure

from that of the prototype AH-64A, [illegible] degrees to partially unload the tail rotor in high speed flight and minimize its flapping motion for improved fatigue life. Figure 15 shows this incidence change. Note that the front face of the spar remains unchanged while the rear face is moved to the left. This keeps the gear box attach points in the same location as the metal baseline, allowing all driveshaft and control hardware to be interchangeable. The leading edge, trailing edge, and tip closure fairings are the same as the baseline except where contours are impacted in a minor way by the four-degree vertical tail incidence change. Steps and hand-hold locations are the same as the metal baseline. See Figure 15.

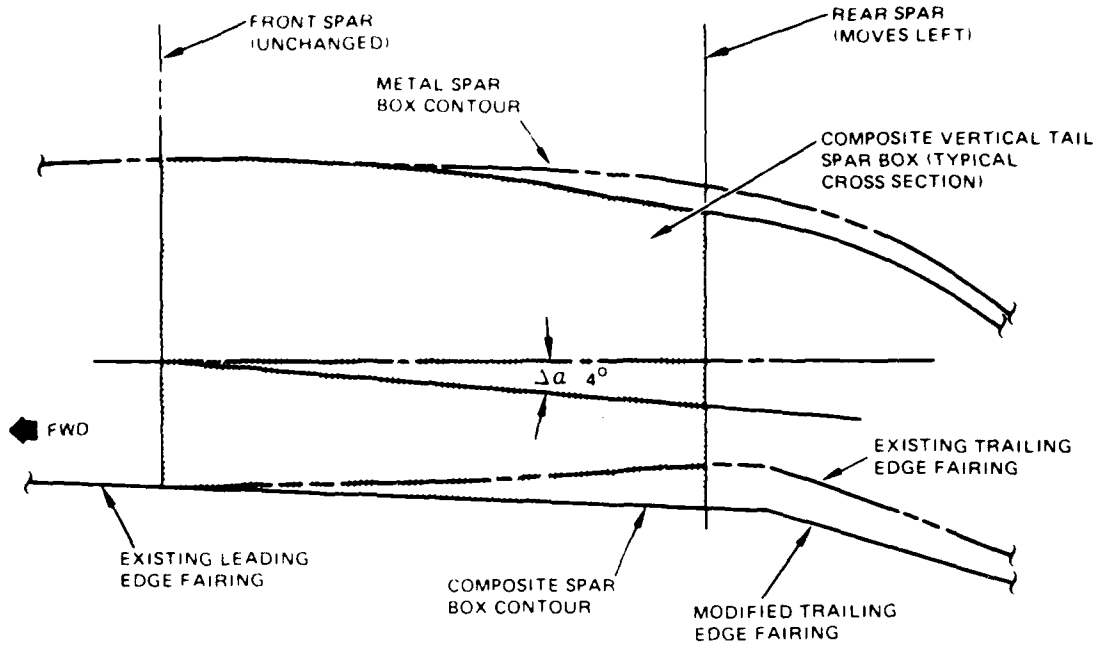
The composite vertical tail is composed of the following components (see Figures 97 and 98):

- Spar box
- Root Fittings
- Upper Gearbox Support Assembly
- Lower Gearbox Support Frames
- Upper Closure Rib
- Steps and Handholds

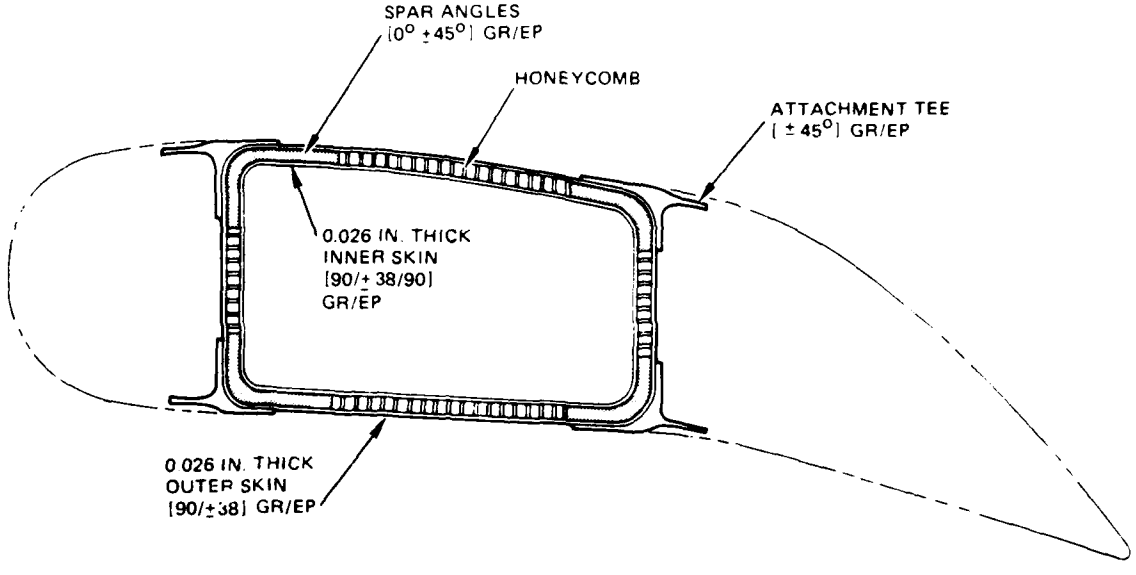
Hardpoints for attaching hardware are provided for the following:

- Tip Fairing
- Leading Edge
- Trailing Edge
- Tail Rotor Brace (2)

The basic structure of the box is a graphite/Kevlar/epoxy filament wound honeycomb sandwich. The skins are ±38-degree helical windings over a 0.250 inch thick Nomex honeycomb core. Graphite/epoxy spar angles (0° ±45° plies) are incorporated in the four corners between the inner and outer skins. The helical windings provide the necessary torsional strength and stiffness, with the corner spar angles supplying bending strength and stiffness. See the lower portion of Figure 16.



VERTICAL STABILIZER INCIDENCE ANGLE MODIFICATION



LEADING AND TRAILING EDGE FAIRING ATTACHMENT TEES

Figure 16. Vertical Tail Cross Sections

The honeycomb sandwich is a balanced structure with each skin 0.026 inch thick. The skins converge to form solid laminates at the root fitting attach region and at the tip end just below the tip cap. Local skin thickness is sized to accommodate the local load requirements. The skin thickness, including laminae and honeycomb core is illustrated in Figure 17. In the center of the spar the sandwich is 0.301 inch thick. The ends taper to 0.060 inch solid laminate.

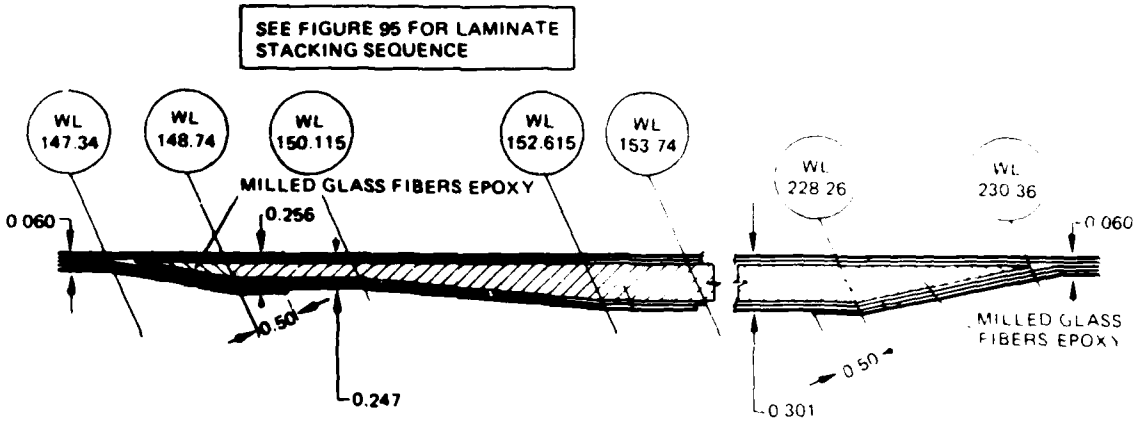


Figure 17. Vertical Stabilizer Skin Thickness

The leading and trailing edge attachment tees are bonded to the box as indicated in Figure 16.

The materials used to fabricate the spar box are:

- Graphite Roving: T300, 3000 filament count
- Kevlar Roving: Kevlar-49, 1420 Denier
- Graphite Fabric: T300, 3000 filament count, eight harness, satin weave
- Epoxy resin: APCO 2434/2347
- Core Adhesive: FM 123-5 film adhesive
- Honeycomb Core: HRH-10 -1/4, 0.250 inch thick

The composite vertical tail attaches to the tailboom with the four-piece aluminum fitting shown in Figure 18. This split fitting attaches to the composite box with HiLok fasteners and carries the bosses for the four 5/8-inch diameter attach bolts. The forward portions of the root fitting also incorporates the intermediate gearbox attach bosses. The root fitting is made from 7049-T6 aluminum alloy.

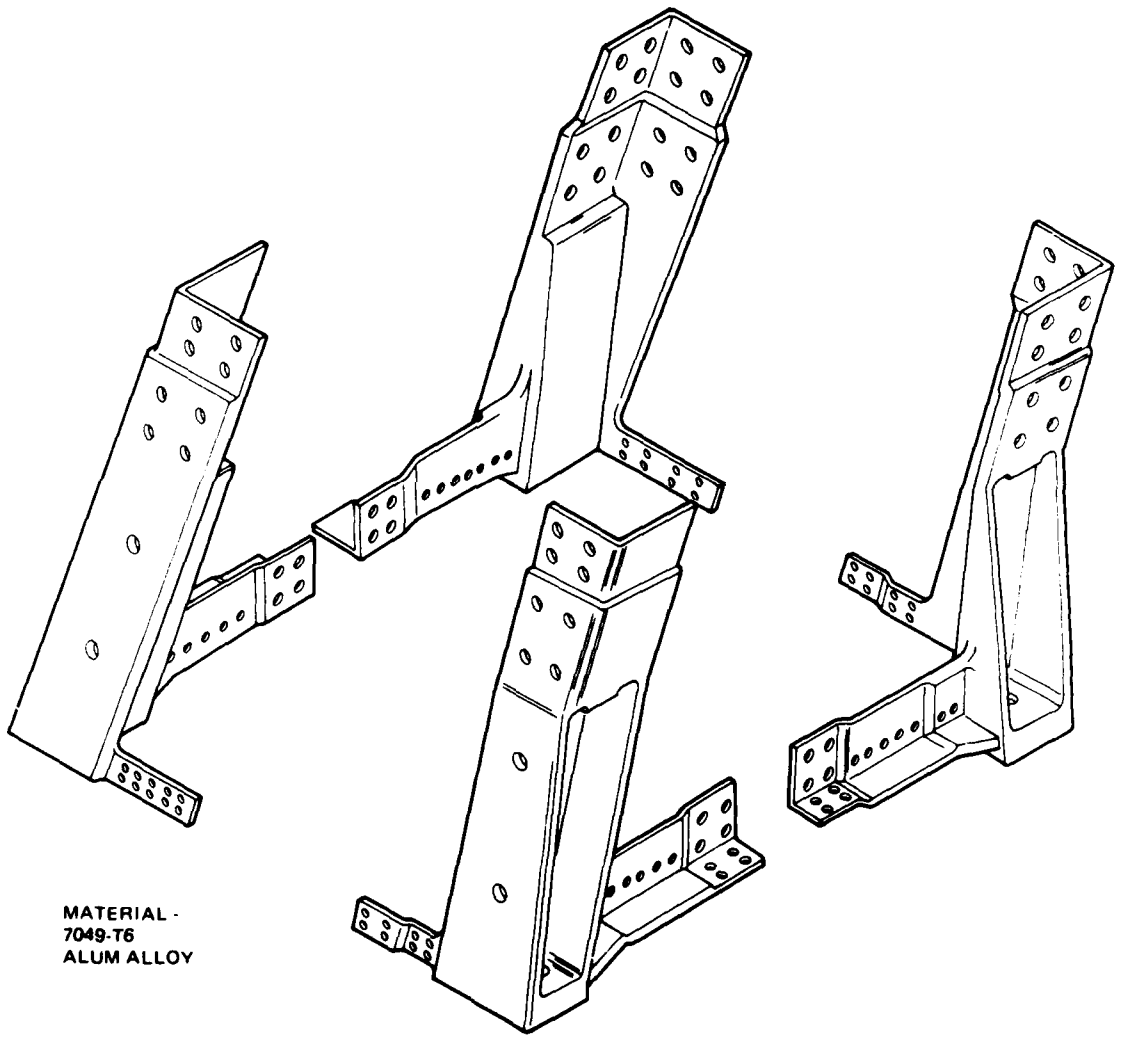


Figure 18. Root Fitting

The upper gearbox mount shown in Figure 19 consists of three graphite/epoxy ribs that have aluminum reinforcing bars bonded in. Self-locking anchor nuts mount the gearbox. The three ribs are connected with a strap to form a prefabricated unit that is assembled with the winding mandrel and wound into the spar box structure. HiLok fasteners are also used to anchor these ribs to the spar box. The lower gearbox supporting rib (Figure 15) is a pre-molded/graphite/epoxy rib that is bolted in place with HiLok fasteners. An aluminum reinforcing bar supports the intermediate gearbox.

The materials used for the ribs are:

•	Rib Assemblies:	Graphite/epoxy:	T300, 3000 filament, eight harness, satin weave, preimpregnated
		E-glass/Epoxy:	1581 glass cloth, preimpregnated
		Adhesive:	FM 123-5
•	Inserts:	2024-T6:	Aluminum alloy

The upper closure is made from graphite/epoxy - T-300, 3000 filament count, eight harness, satin weave, preimpregnated.

The steps and handhold are graphite/epoxy tubes fixed on the vertical tail spar box as shown in Figure 15. That portion extending into the air stream is elliptically shaped for reduced drag (See Figure 20).

The materials for the steps are:

- Graphite/epoxy: T-300, 3000 Filament count, eight harness, satin weave, preimpregnated
- Graphite roving: T-300, 3000 filament count
- Epoxy resin: APCO 2434/2347

The leading and trailing edge fairings attach to flanges that are bonded to the corners of the spar box. These are T-section flanges as indicated in Figure 16. They are made from:

- Graphite/epoxy: T-300, 3000 Filament Count, eight harness, satin weave, preimpregnated

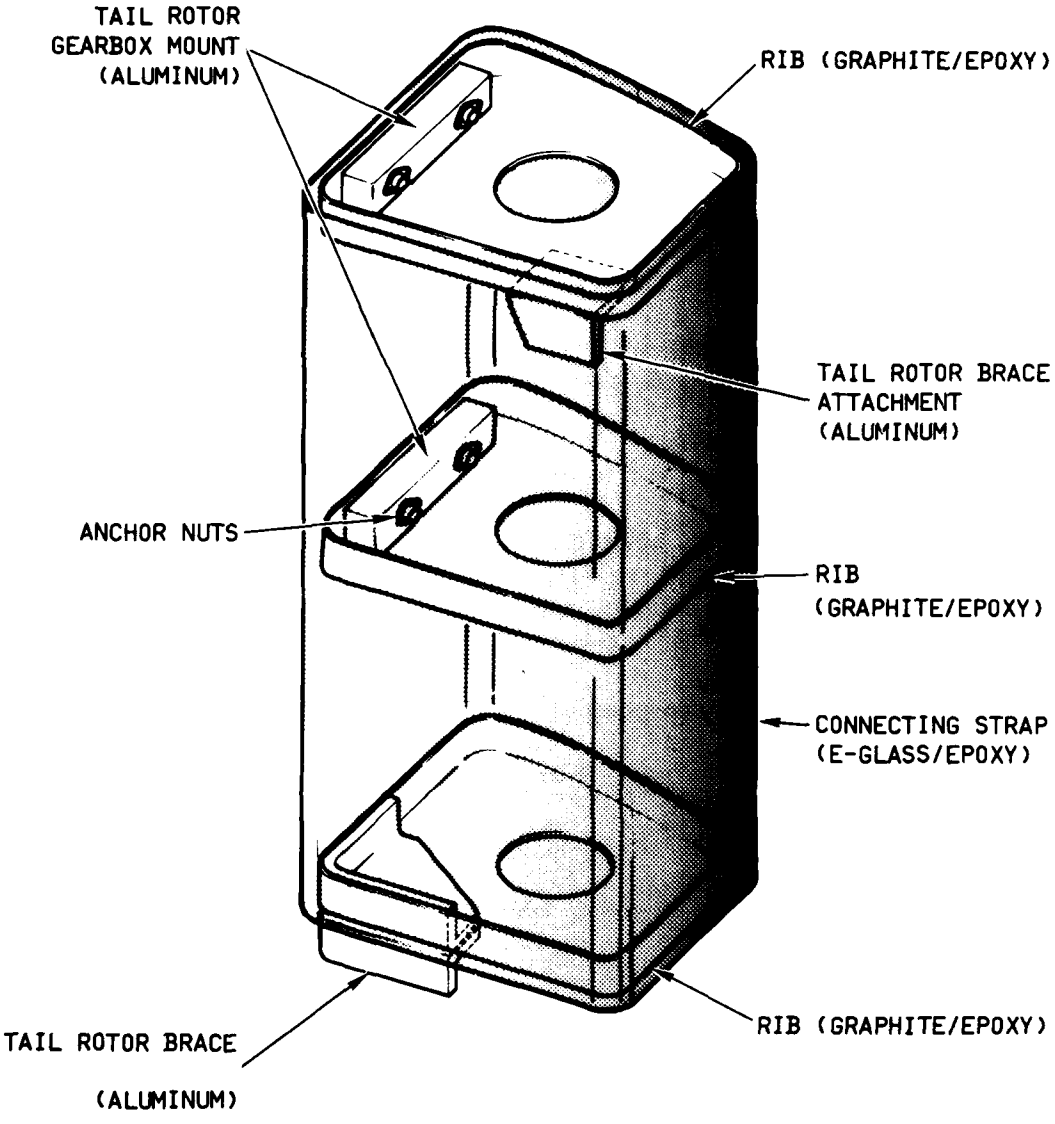


Figure 19. Upper Gearbox Assembly

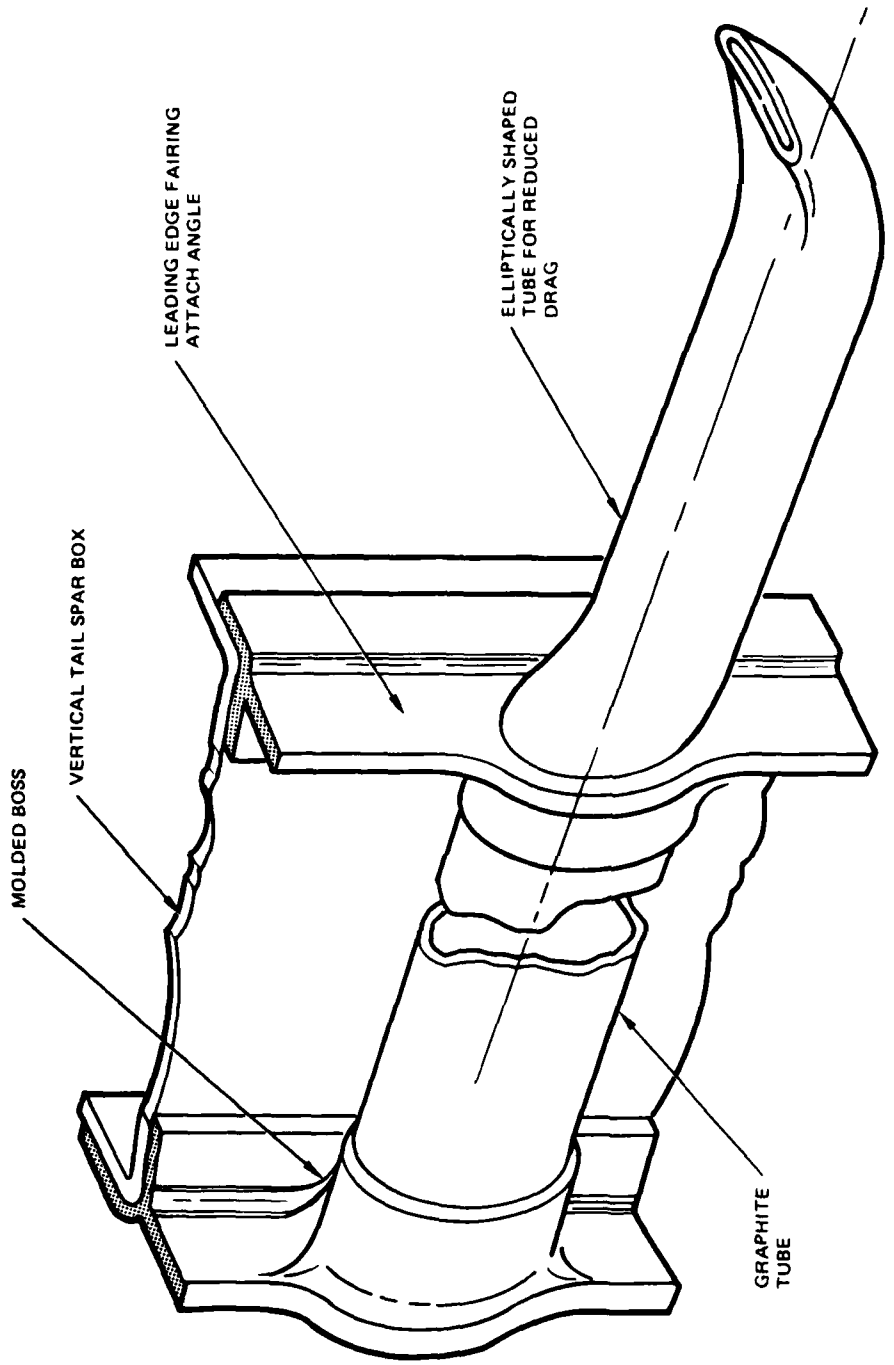


Figure 20. Step/Hand Hold

STABILATOR

The composite stabilator's geometry is:

Area - ft^2	33.36
Span - in.	128.0
Tip chord - in.	31.85
Root chord - (at centerline) - in.	43.2
Airfoil	NACA 0018
Aspect ratio	3.41
Chordline for zero sweepback, percent	50

The composite stabilator is shown in Figure 21. It has the same outside contour as the AH-64A metal stabilator, but with added leading edge sweepback. The pivot point and actuator locations are changed to facilitate hinging the stabilator to the tailboom instead of to the vertical tail. This change permits decoupling of the vertical tail/stabilator dynamic responses and also allows the hinge spacing to be increased, thus reducing the hinge loads. An improved actuation system replaces the prototype helicopter's tandem electric actuator by a pair of side-by-side hydraulic actuators attached to the tailboom aft bulkhead. The hinge is also part of the aft bulkhead.

The stabilator is made up of the following structural members (see Figure 99):

- Skins (left, right, and center)
- Spars (forward and aft)
- Ribs (left and right, 3 each)
- Nose ribs (left and right, 3 each)
- Reinforcing ribs (left and right, 1 each)
- Tip closures (2)

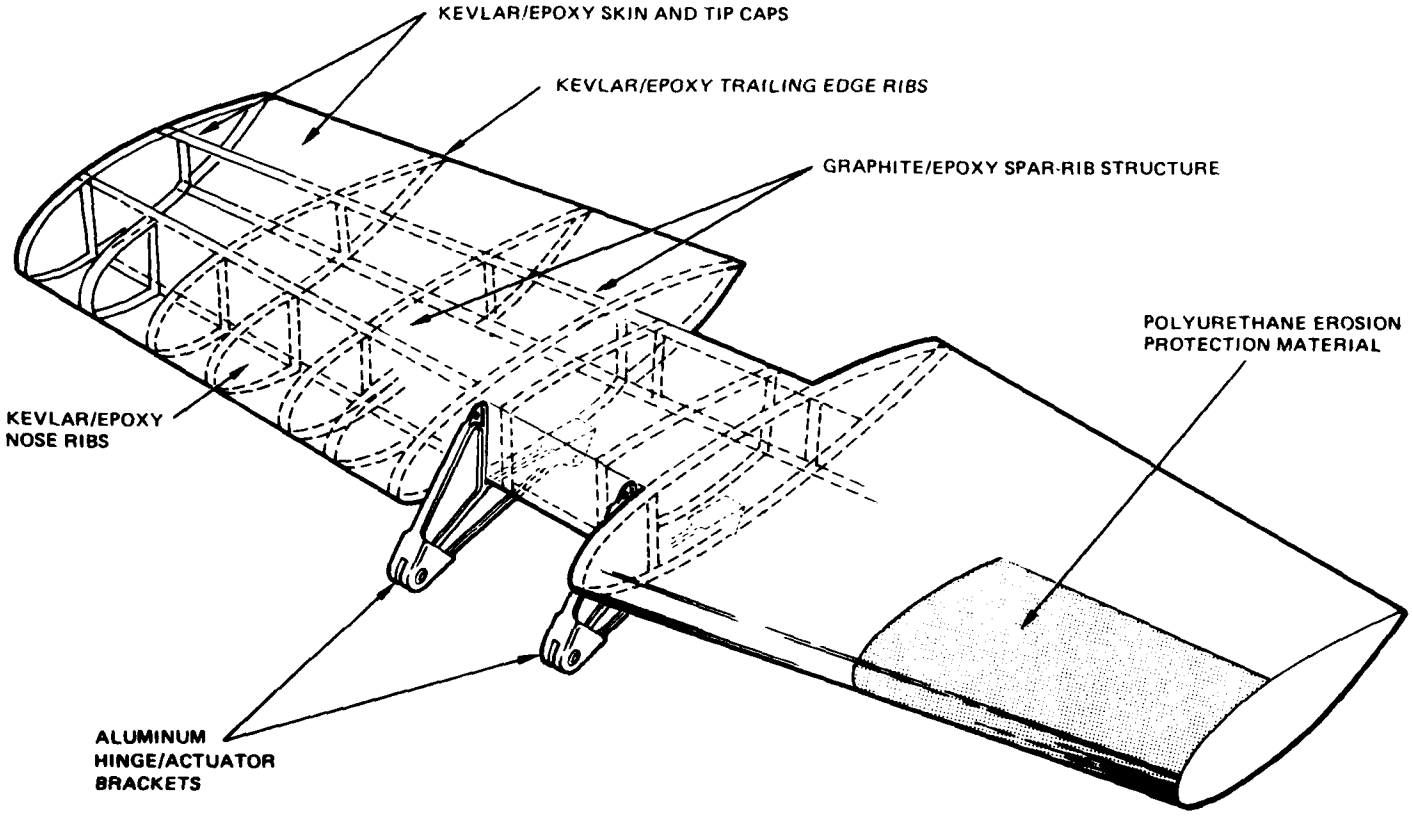


Figure 21. Stabilator Structure

Attach details are:

- Pivot/actuator fitting (left and right)
- Bathtub fitting (left and right)

In addition, the stabilator is protected from rocket debris by a polyurethane cover.

The skin is made of four pieces, (left, right, upper center, lower center). The skin provides the torsional stiffness and shear tie for the spars. The spars provide axial bending stiffness and resist air loads and diagonal tensions loads caused by thin skin shear buckling. The left and right-hand skins are premolded to the airfoil shape, fit over the spar/rib structure, and are bonded to it. The skin is reinforced in the center section for torsional requirements. The total skin thickness is 0.030 inch with an additional 0.030 inch thickness in the center box area between the root end ribs.

The skin material is:

- Kevlar roving: Kevlar-49 roving, 1420 Denier
- Epoxy Resin: APCO 2434/2347

The forward and aft spars consist of C-section beams. Spanwise bending stiffness is provided by unidirectional fiber in the caps oriented in the spanwise direction. The spars are made of:

- Graphite roving: T300, 3000 Filament Count
- Epoxy Resin: APCO 2434/2347

The ribs are molded as three element boxes with flanges all around their peripheries. The portions of the ribs that lie between the spare are graphite/epoxy, while the leading and trailing edge portions are Kevlar/epoxy. The materials used to make the ribs are:

- Graphite Roving: T300, 3000 filament count
- Kevlar Roving: Kevlar-49 Roving, 1420 Denier
- Epoxy Resin: APCO 2434/2347

The tip closures provide a shear rib at the outboard end of the airfoil and serve to shield the edges of the skin and the substructure from environmental effects. The material is the same Kevlar/epoxy used for the skins.

The pivot/actuator fittings are two aluminum alloy fittings attached to the stabilator assembly with stainless steel bolts. The pivot holes are bushed with TFE-lined stainless steel bushings. The materials for these fittings that are shown inFigure 21 are:

- Fitting: 7049-T411 aluminum alloy
- Bushing: CRES steel, TFE coated

Rocket debris protection is provided by a polyurethane cover 0.050 inch thick extending over the outboard 23.5 inches of the span on both sides. This material wraps around the leading edge and covers the forward five inches of the upper surface and all but the aft six inches of the lower surface.

ENVIRONMENTAL PROTECTION

Extensive investigation during the advanced composites programs has established that there will be no corrosion problem with the composite components. However, suppression of galvanic corrosion between aluminum fittings and adjacent graphite/epoxy laminates requires the use of a layer of E-glass/epoxy for electrical isolation at all interfaces.

All steel parts are cadmium plated and all aluminum parts are chromic acid anodized. All fasteners are corrosion-resistant steel. Final painting consists of a urethane top coat over a suitable primer for protection against ultraviolet radiation.

Figure 22 shows how lightning protection is provided by a grounding system that extends from a lightning collector in the tip cap of the vertical tail, down through the aluminum ground plane for the antennas in the trailing edge fairing of the vertical tail, and out through the tailboom's aluminum frame at Sta 547 and the tail landing gear strut. A grid of aluminum foil around the leading and trailing edges and tip to two discharge wicks on each stabilator tip and through a jumper into the aluminum rear frame in the tailboom provides lightning protection for the stabilator.

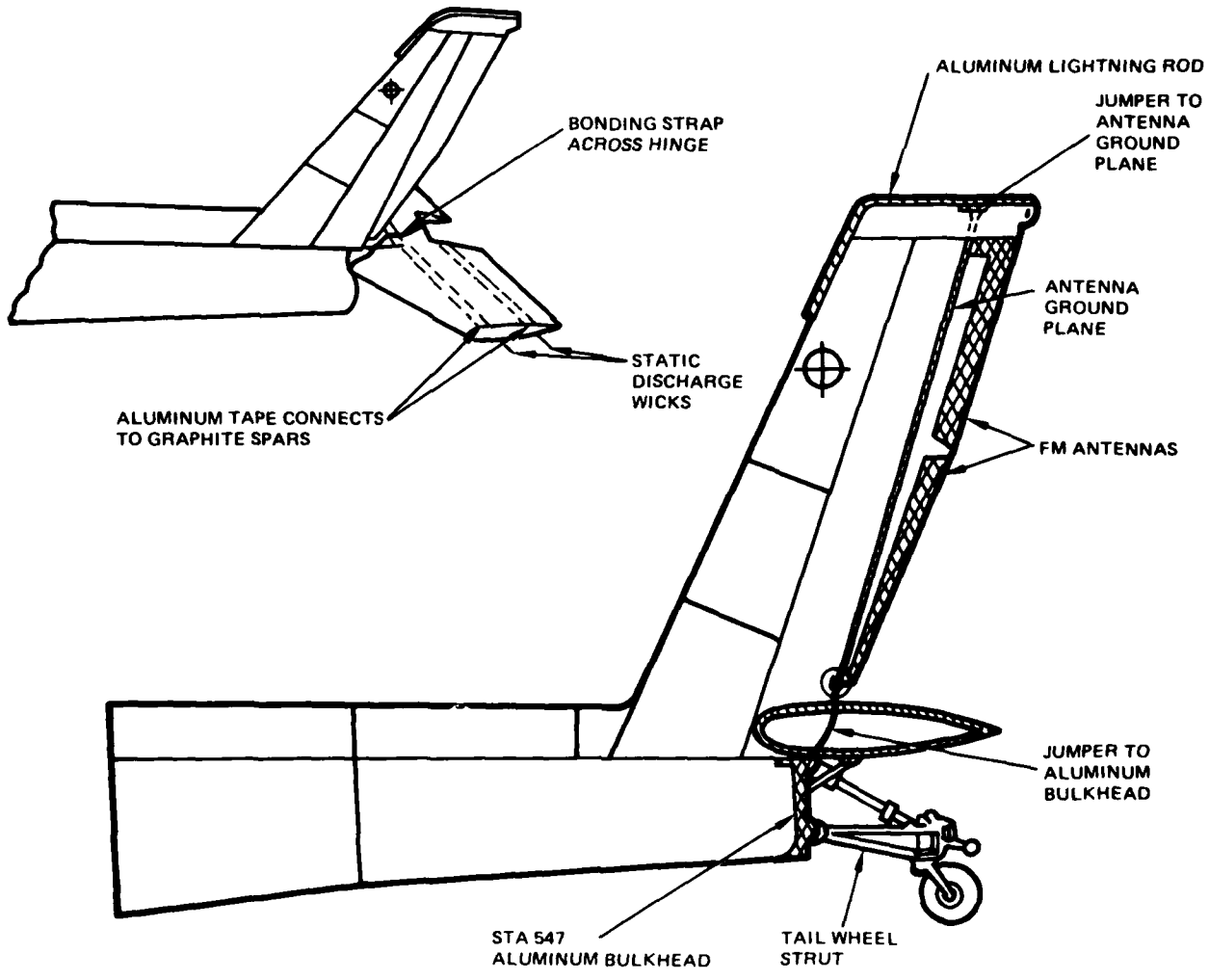


Figure 22. Lightning Protection

WEIGHT ANALYSIS

The weight analysis for the CTS is given in detail in Table 2 per MIL-STD-1374 Part III for the three components -- stabilator, vertical tail, and tailboom. The calculations for these weights are based on released drawings (84 percent by weight), and on pre-release drawings for the remaining details and installation. The component weights are summarized in Table 3 and are compared with their counterparts in the basic metal helicopter. The weight saving is shown to be 72.4 pounds.

Two adjustments are needed. First, a calculation for a preliminary layout of the metal-to-composite splice joint at Sta 370 shows an incremental weight increase of 7.6 pounds relative to a structure without a manufacturing break at this point. Second, the CTS is designed to accept the Composite Flexbeam Tail Rotor (CFTR) that increases the load on the tailboom by six percent relative to the metal tail rotor. A weight analysis of a strengthened metal tailboom to accept this load indicates a weight increase of 6.0 pounds is needed. Hence, the total adjustment for the Sta 370 splice and the CFTR yields an overall weight reduction of 70.8 pounds for the CTS versus the metal tail section modified to accept the CFTR.

The longitudinal center of gravity for the AH-64A equipped with the CTS is given in Figure 23. For comparison, the normal envelope portion of the chart for the helicopter with the metal tail section is shown as a dotted line in Figure 23. Note that the lighter weight CTS results in a more forward center of gravity for the vehicle.

TABLE 2. CTS WEIGHT ANALYSIS

	Original Goal	Current Status	Basis For Current Data	
			% EST	% CALC
Stabilator - Basic	(58.0)	(52.0)	(18)	(82)
Upper - Cover	3.0	3.2	0	100
Lower - Cover	3.0	3.2	0	100
Spars	7.0	15.2	0	100
Interspar - Ribs	4.0	6.0	0	100
- Chordwise Stiff.	0.0	0.0	0	0
Leading Edge - Cover	7.0	3.6	0	100
- Ribs	2.0	1.4	0	100
- Fasteners	1.0	0.0	0	0
Trailing Edge - Cover	14.0	3.6	0	100
- Ribs	4.0	0.6	0	100
- Fasteners	2.0	0.0	0	0
Fitting - Hinge (pivot)	5.0	8.0	100	0
- Actuator	2.0	1.0	0	100
Attachments - Misc. & Fasteners	2.0	2.3	50	50
Tips	2.0	0.0	0	0
Debris Protection	0.0	3.8	0	100
Lightning Protection	0.0	0.1	0	100
Vertical Tail - Basic	(79.0)	(90.0)	(22)	(76)
Cover	26.0	0.0	0	0
Spar Webs	21.0	0.0	0	0
Spar Box Assembly	0.0	34.2	0	100
Interspar Ribs	10.0	9.5	26	74
Leading Edge - Cover	7.0	11.5	0	100
- Ribs	2.0	2.2	0	100
- Fasteners	1.0	2.5	0	100
Trailing Edge - Cover	3.0	4.3	0	100
- Ribs	1.0	0.6	0	100
- Fasteners	0.0	2.1	0	100
Attachments - Tail to Fuselage	2.0	12.0	100	0
- Miscellaneous	1.0	3.0	100	0
Tips	3.0	3.5	0	100
Steps & Grips	2.0	2.0	100	0
Tail Rotor Gear Box Fittings	0.0	2.6	0	100
Tailboom - Basic	(117.6)	(139.5)	(12)	(88)
Bulkheads + Frames	16.0	27.2	48	52
Joints, Splices + Fasteners	8.6	3.4	70	30
Monocoque Sandwich	89.0	100.4	0	100
Intercostals	0.0	4.4	0	100
Stabilator Actuator Fitting	2.0	0.0	0	0
Jack Fitting	2.0	1.2	100	0
Steps + Grips	0.0	1.4	0	100
Tail Rotor Fairing Angles	0.0	1.5	0	100

TABLE 3. CTS WEIGHT STATUS SUMMARY

Item	Metal Weight (lb.)	Composite Weight (lb.)	Weight Savings (lb.)
Stabilator	73.1	52.0	21.1
Vertical Tail	120.0	90.0	30.0
Tailboom	160.8	139.5	21.3
Totals	353.9	281.5	72.4
Estimated increase in weight at fuselage station 370 to install CTS (reduced weight savings)			- 7.6
Net Weight Savings			64.8
Estimated weight increase of metal tailboom to accommodate CFTR			+ 6.0
Adjusted Weight Savings			70.8

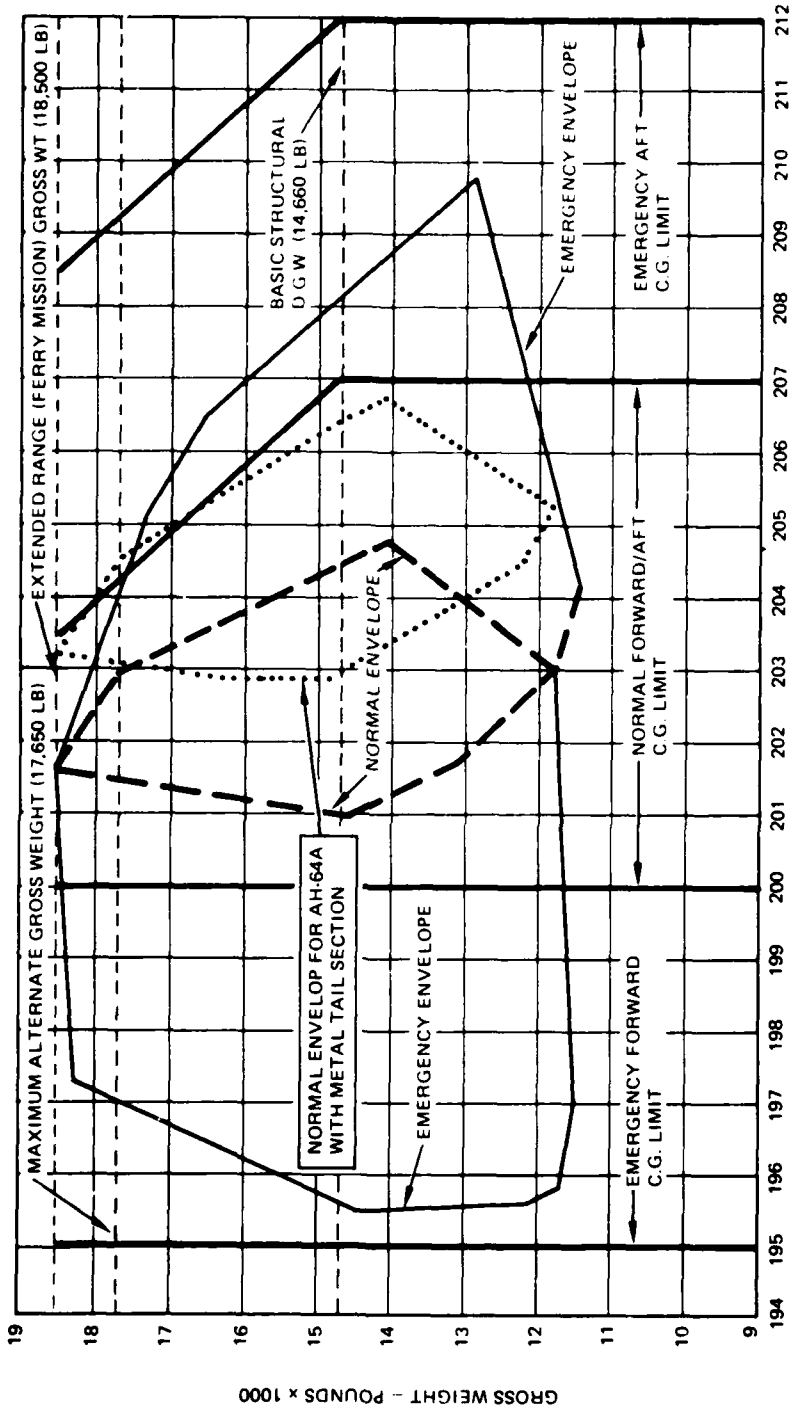


Figure 23. AH-64A Longitudinal Center-of-Gravity with CTS

RELIABILITY ASSESSMENT

Reliability assessment of the CTS is based on the results of a preliminary systemized Failure Modes, Effects, and Criticality Analysis (FMECA), a review of program requirements, and a review of reliability studies conducted for other composite programs at HHI. The analysis considers the degradation of reliability that may occur as a result of manufacturing defects, storage, environmental conditions, in-service conditions, and production techniques. Specific areas of concern include delamination and disbonding, porosity and voids, resin rich and resin starved areas, internal geometry and bond lines, resistance to impact, and defects in metal attach fittings. These studies show that the hazards imposed by air vehicle operation and maintenance are the primary reliability considerations. Detection through such nondestructive inspection techniques as X-ray or Bondoscope permits these problems to be detected and cured.

Cracking occurs primarily at fastener interfaces and is easily detectable. The CTS is designed to use a minimum of fasteners and the composite structure is adequately designed to accept fasteners. Degradation due to limited cracking is expected to be minor.

Resin-starved/resin-rich areas can contribute to structural degradation and loss of effective performance. Resin starvation can fail to support the filaments properly and result in delamination; this can result in geometric misplacement of the fibers to cause unanticipated load paths. Resin richness means either that not enough filaments are present to carry the loads or that the resin/filament matrix is oversized and, hence, out of its desired position in the structure -- in either case structural integrity is jeopardized. These conditions occur only during manufacturing not during in-service operation and are discovered during nondestructive evaluation. A resin-rich or resin-starved defect is not considered to be a failure mode in the FMECA because it is a failure mechanism (cause) of a delamination failure mode.

Ultraviolet light, rain erosion, moisture entrapment, extreme temperatures, snow, wind, and lightning contribute greatly to reliability degradation. The resin matrix material degrades considerably when exposed to ultraviolet light. Protection from ultraviolet light is provided by a suitable coating of the exposed skins. Although moisture entrapment cannot be avoided, the composite tail section is designed for the worst case condition as is true with the rest of the environmental extremes. Extremely rare incidents such as heavy hail or hurricane velocity winds are not considered since these environments should be avoided when operating the air vehicle.

The significant contributors to reliability degradation of the CTS are hazards induced during air vehicle operation and maintenance. These include thermal cycling, shock, vibration, aircraft fluids, rotor downwash (induced airborne particles and FOD), rocket debris, rough handling, impact with terrain objects, maintenance, and contact with work stands and ground vehicles. Design allowables may compensate for some of these hazards and the resultant degradation may be readily visible. However, the resultant degradation of those hazards, not readily visible, can only be determined by an effective and adequate nondestructive evaluation or nondestructive test technique. Based on results of previous testing and for equivalent material thickness, the order of damage tolerance is as follows. Fiberglass is the most damage tolerant, Kevlar is next best, and graphite has low damage tolerance. Sandwich construction has poorer impact resistance than monolithic construction and tends to suffer reductions in strength due to subsurface damage. Nomex sandwich construction is protected by Kevlar/graphite facing sheets on the composite tail section and Nomex has shown to have better damage tolerance than aluminum. Finally, simple field-type repair methods have been shown to be effective for many types of routine impact damage. At the time that the program was terminated, a Failure Modes, Effects, and Criticality Analysis (FMECA) had not been made for comparison with the metal empennage of the basic AH-64A helicopter.

STRUCTURE VERIFICATION TESTS

Tests were made to determine the properties of the particular laminates used in the tailboom, vertical stabilizer, and stabilator of the CTS. The testing included tension, compression, shear, and bearing of typical configurations found in the CTS to verify strength, elasticity, and strain allowables.

Two types of test specimens were investigated: helically wound tubular specimens used for tension, compression, and torsional shear tests and flat panel specimens laminated from wet-filament-wound broadgoods with fabric reinforcing plies. Test fixture end reinforcement was applied to the wet specimens at each end. Specimen identification, material description, and type of test as well as number of specimens are presented for tubular tension and compression specimens in Figure 24 and Table 4, for tubular torsion specimens in Figure 25 and Table 5, and for bolt bearing test specimens in Figures 26 and 27 and Tables 6 and 7. All of the specimens were wet-filament wound using Kevlar-49 and Thornel 300 graphite fibers with APCO 2434/2347 (7.5 PPHR) resin. The fiber volume ratio was 0.50 for all specimens.

All specimens were tested in a room temperature (70°- 75°F), dry condition. Each tension and compression tubular specimen was installed in a test machine as shown in Figures 28 and 29 in a manner to prevent any bending loads from being applied. (A stabilizing bar was installed in the compression specimens to prevent buckling.) A two-inch extensometer was mounted to each specimen to measure deflection. A load-deflection strip-chart recorder monitored the applied load and deflection. The load-deflection curve was used to calculate the modulus of elasticity of the specimen. The load was applied at an average rate of 0.05 inches per minute for the moving head of the test machine. All specimens were tested to failure.

Each tubular torsion test specimen was installed in a test fixture designed to apply shear loads to the test section of the specimen as shown in Figure 30 with a stabilizing bar installed to prevent buckling. Holes were drilled in the end pieces of the specimens to firmly attach them in the test fixture.

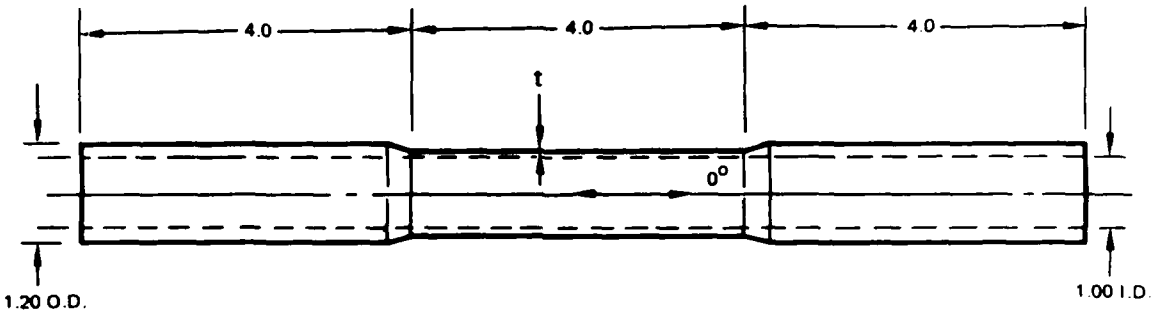


Figure 24. Tension/Compression Specimen

TABLE 4. TENSION/COMPRESSION SPECIMENS

Specimen No.	Component	Layup Composition	Total Thickness	Condition
I-1	Stabilator Skin	[±45°/0°/±45°] Kevlar (KV)	.020/.020/.020 .060	Tension Compression
I-2	Vertical Tail Skin	[±30°/90°/±30°] Graphite (GR)	.020/.020/.020 .060	T C
I-3	Stabilator & Vertical Tail Spar Caps	$\pm 45°/0°_{12}/\pm 45°$ GR	.014/.085/.014 .113	T C
I-4	Tailboom	$\pm 45°/0°_{12}/\pm 45°$ GR/KV 50-50	.014/.083/.014 .111	T C
I-5	Tailboom	$[90°, (\pm 13.5°)_2, 90°]$ GR/KV 50-50	.007/.038/.007 .052	T C
I-6	Tailboom	$[90° (\pm 8°)_2 90°]$ GR/KV 50-50	.007/.032/.007 .046	T C

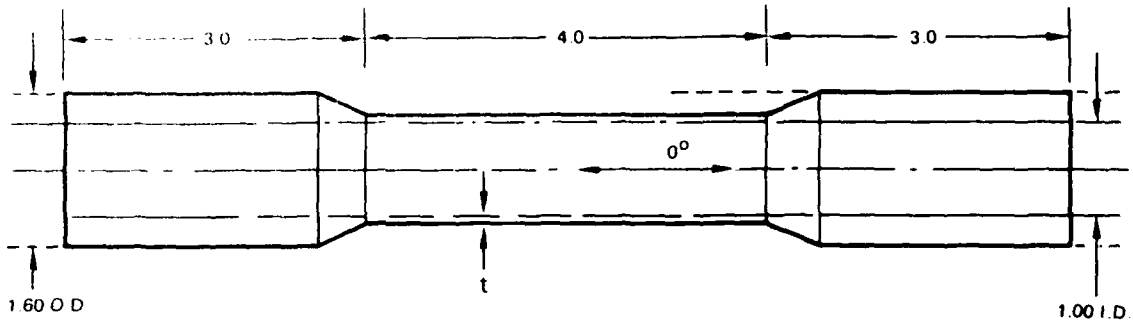
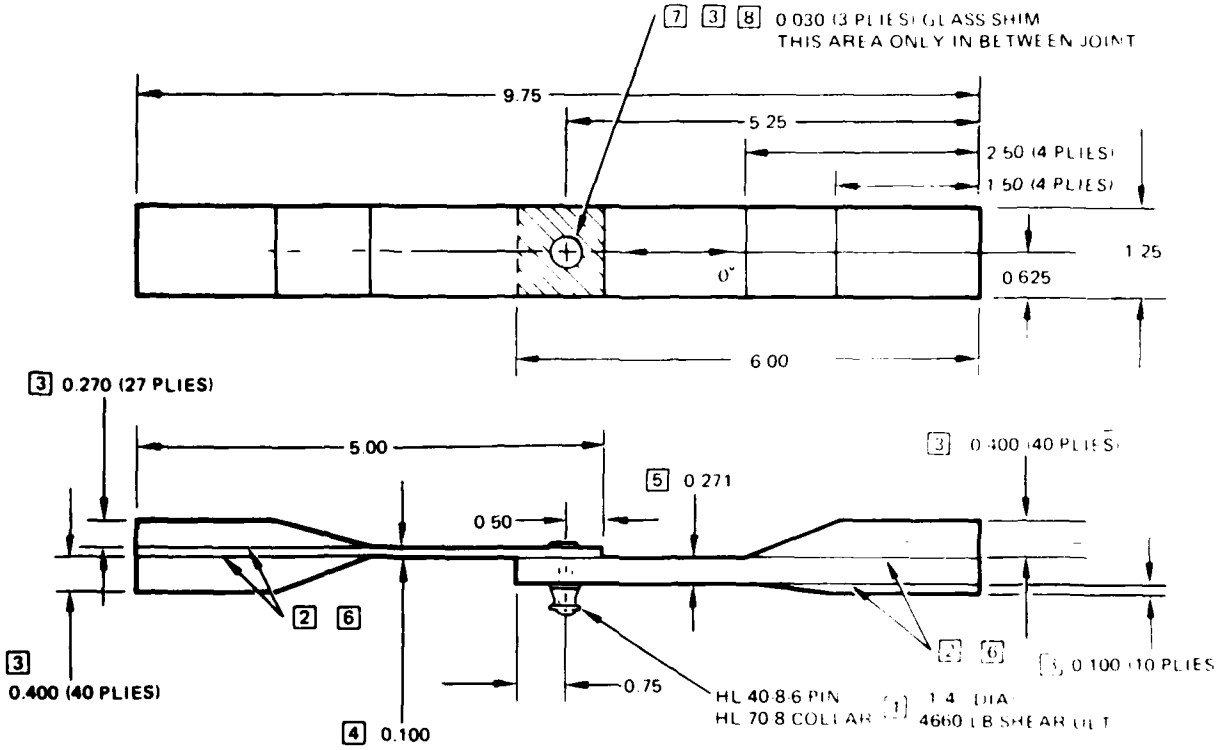


Figure 25. Torsion Test Specimen

TABLE 5. TORSION SPECIMEN

Specimen No.	Component	Layup Composition	Skin Thickness
II-1	Stabilator Skin	[±45°/0°/±45°] KV	.020/.020/.020 .060
II-2	Vertical Tail Skin	[±30°/90°/±30°] GR	.020/.020/.020 .060
II-3	Tailboom	[90°, $(13.5°)_2$, 90°] GR/KV 50-50	.0066/.0382/.0066 .051
II-4	Tailboom	[90°, $(\pm 8°)_2$, 90°] GR/KV 50-50	.0066/.0320/.0066 .0452
II-5	Tailboom	[90°, $(\pm 9°)_2$, 90°] GR	.010/.032/.010 .052
II-6	Tailboom	[90°, $(\pm 15°)_2$, 90°] GR	.010/.034/.010 .054



[8] LAMINATE PER HP 15-42

[7] APCO 2434-2347 EPOXY RESIN

[6] EA 934 NA ADHESIVE ROOM TEMP CURE

[5] FOR LAMINATE MATERIAL CALLOUT AND STACKING SEQUENCE SEE TABLES 6 AND 7

[4] 4340 OR 4130 STEEL PLATE (0.100 x 1.25 x 5.00. HEAT TREAT 150 KSI UTS

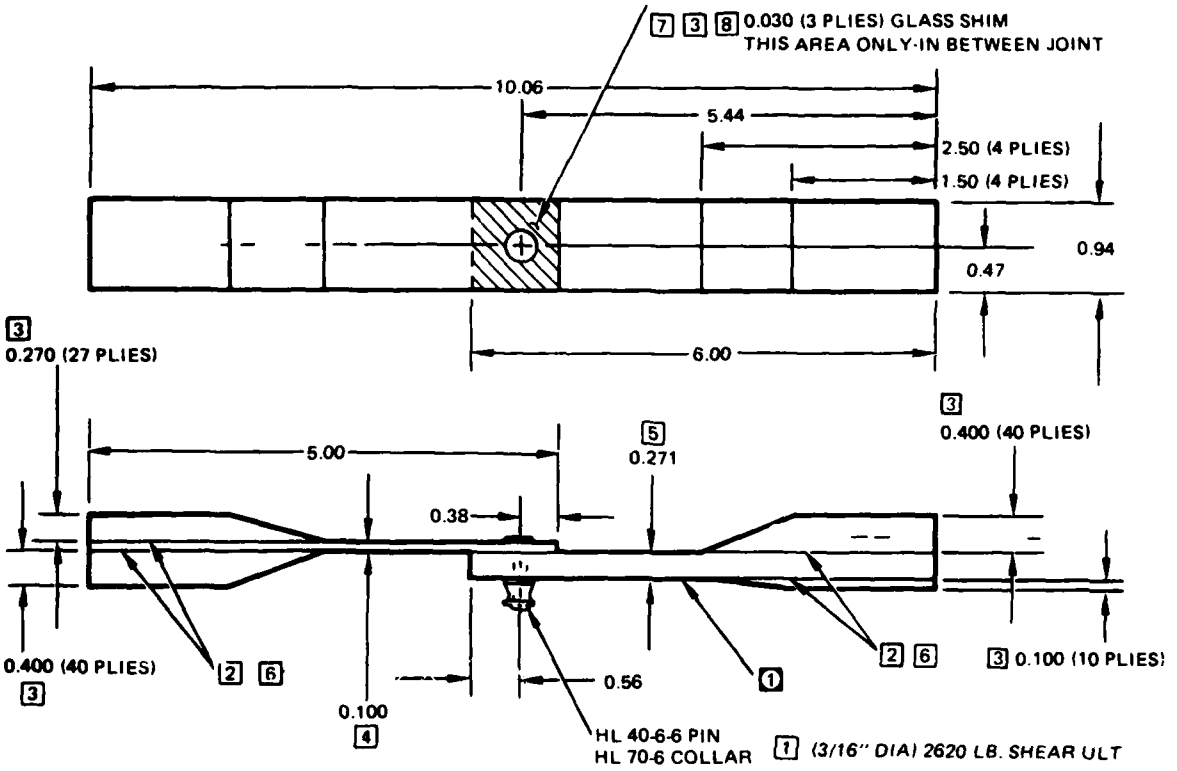
[3] E-GLASS FABRIC MIL-C-9084 1581 CLOTH TYPE VIII, CL2, GR A(0.010/PLY)

[2] SAND BLAST BONDING SURFACES PRIOR TO BONDING

[1] INSTALL HI-LOK FASTENERS PER HP 3-2

NOTES:

Figure 26. Bearing Test Specimen Type III-1



[8] LAMINATE PER HP 15-42

[7] APCO 2434-2347 EPOXY RESIN

[6] EA 934 ADHESIVE ROOM TEMP CURE

[5] FOR LAMINATE MATERIAL CALLOUT AND STACKING SEQUENCE SEE TABLE, 6 AND 7

[4] 4340 OR 4130 STEEL PLATE (0.100 x 0.94 x 5.00) HEAT TREAT 150 KSI UTS

[3] E-GLASS FABRIC MIL-C-9084 1581 CLOTH TYPE VIII, CL 2, GR A (0.010/PLY)

[2] SAND BLAST BONDING SURFACES PRIOR TO BONDING

[1] INSTALL HI-LOK FASTENERS PER HP 3-2

NOTES:

Figure 27. Bearing Test Specimen Type III-2.

TABLE 6. STACKING SEQUENCE FOR TYPE III SPECIMENS

Ply Number	Angle	Material*	Ply Thickness
P15	90°	⑥ ⑧	.011
P14	±45°	④	.013
P13	±8°	⑤ ⑥	.022
P12	±45°	④	.013
P11	90°	⑦	.020
P10	±45°	④	.013
P9	90°	⑦	.020
P8	±45°	④	.026
P7	90°	⑥	.011
P6	±45°	④	.026
P5	90°	⑥	.011
P4	±45°	④	.026
P3	±8°	⑤ ⑥	.022
P2	±45°	④	.026
P1	90°	⑥ ⑧	.011

*See Table 7.

TABLE 7. FABRICATION NOTES FOR TYPE III SPECIMENS

(8)	APCO 2434-2347 Epoxy Resin HMS 16-1115 Type I, CL 1
(7)	E-Glass, Roving MIL-R-60346 Type I, CL 1
(6)	T300 3000 Fil Count, Roving HMS 16-1163 Type I, CL 1
(5)	Kevlar 49, Roving 1420 Denier HMS 16-1164 Type I, CL 3
(4)	T300 3000 Fil Count, Fabric HMS 16-1163 Type II, CL 1, GRA
(3)	Filament wind per HP 15-67
(2)	Layup per HP 15-42
(1)	First Ply goes Against Tool Face

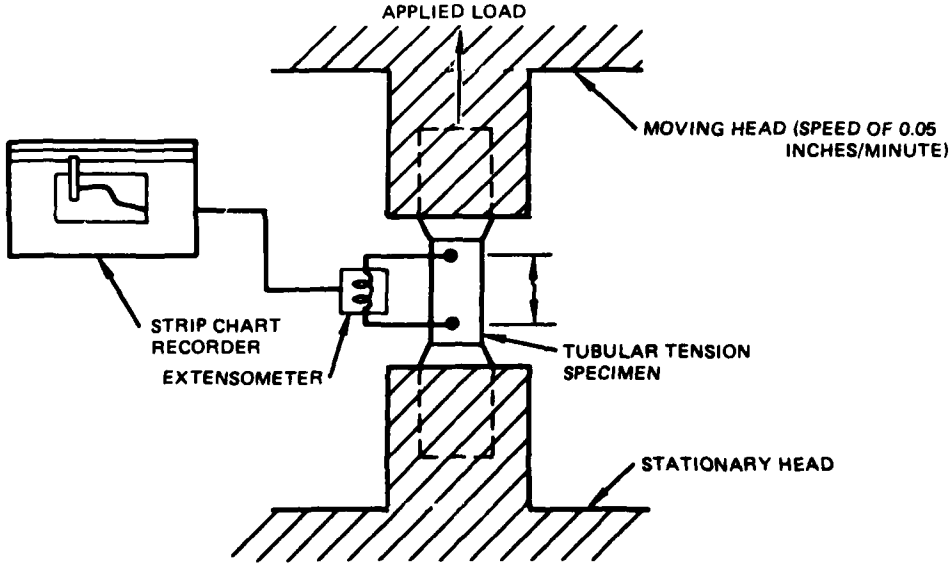


Figure 28. Tubular Static Tension Test Schematic

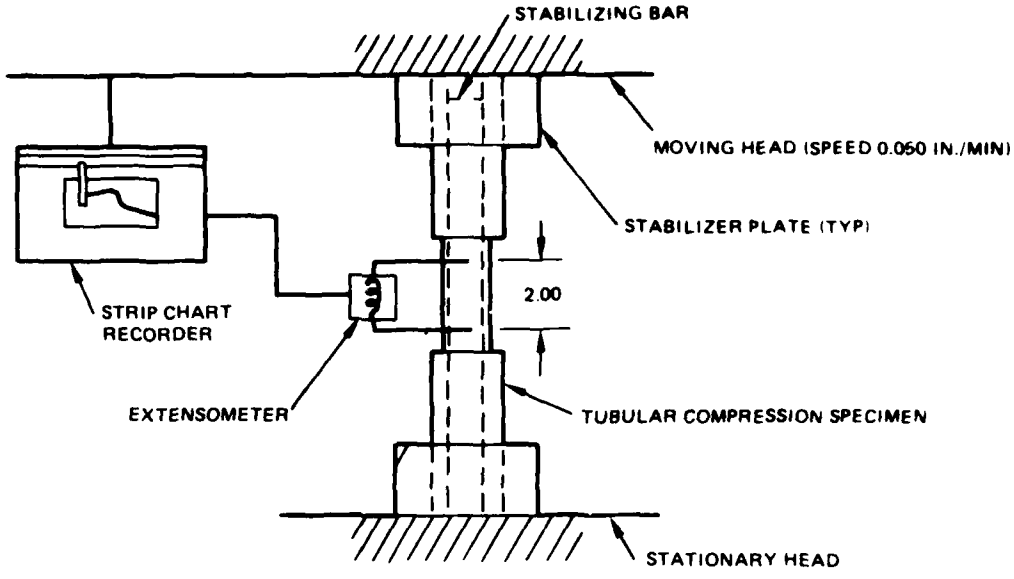


Figure 29. Tubular Static Compression Test Schematic

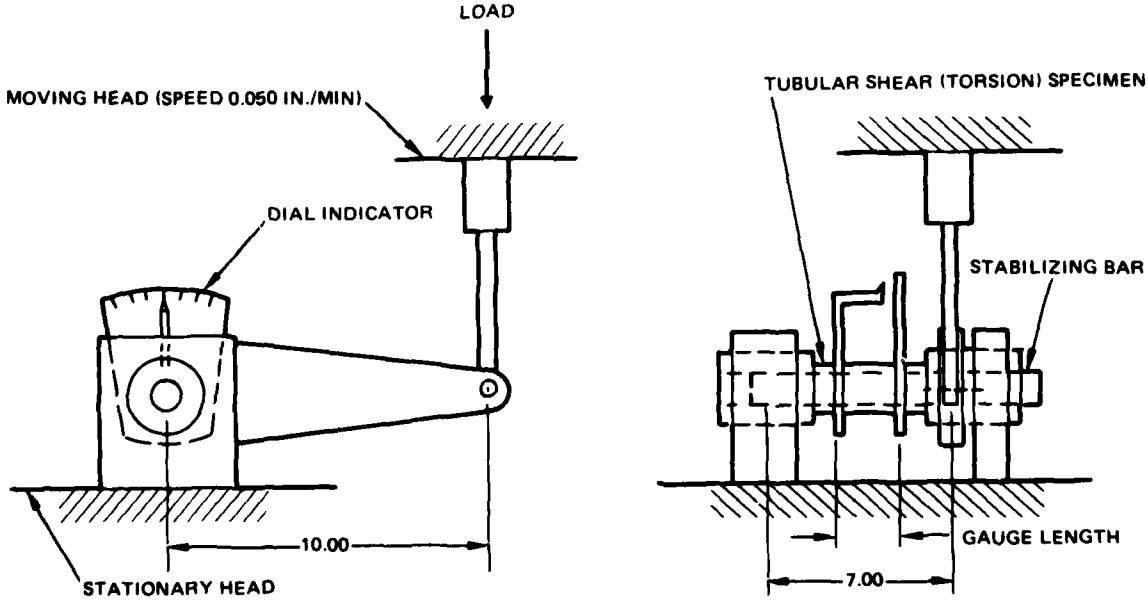


Figure 30. Tubular Static Shear (Torsion) Test Schematic

A gage indicator was mounted to the test section of the specimen to measure angular displacement during testing. A torque arm was mounted to one end of the specimen and attached to the moving head of the test machine while the other end of the test specimen was fastened to the fixed head of the machine. The torque arm was designed to preclude the application of bending loads to the specimen. The load was applied to the torque arm at the rate of 0.05 inches per minute as for the other tests with each specimen loaded to failure. Angular deflection was measured at periodic intervals during the test. The bolt bearing test specimens were tested in a manner similar to the tension specimens except the deflections were not monitored.

A total of 64 specimens were tested. Test results are given in Tables 8, 9 and 10 for tension/compression, torsion, and bearing tests respectively. The dimensions, ultimate load, ultimate stress, and modulus of elasticity for each specimen are given.

TABLE 8. TENSION/COMPRESSION TEST RESULTS

Specimen					Applied Load			
Type[a]	No.	I.D. (in.)	O.D. (in.)	Area (in.²)	Type	Ult. (lb)	Ult. Stress (psi)	[illegible]
I-1	1	0.991	1.109	[illegible]	Tension	[illegible]	[illegible]	[illegible]
	2	0.993	1.106	[illegible]	T	[illegible]	[illegible]	[illegible]
	3	0.993	1.116	[illegible]	T	[illegible]	[illegible]	[illegible]
	4	0.993	1.109	[illegible]	Comp	[illegible]	[illegible]	[illegible]
	5	0.993	1.116	0.204	C	[illegible]	[illegible]	[illegible]
	6	0.992	1.109	[illegible]	C	[illegible]	[illegible]	[illegible]
I-2	7	0.996	1.115	0.197	T	[illegible]	[illegible]	[illegible]
	8	0.995	1.114	0.197	T	[illegible]	[illegible]	[illegible]
	9	0.996	1.111	[illegible]	T	[illegible]	[illegible]	[illegible]
	10	0.996	1.113	[illegible]	C	[illegible]	[illegible]	[illegible]
	11	0.995	1.118	0.204	C	[illegible]	[illegible]	[illegible]
	12	0.994	1.108	[illegible]	C	[illegible]	[illegible]	[illegible]
I-3	15	1.017	1.239	[illegible]	C	[illegible]	[illegible]	[illegible]
	16	1.016	1.237	0.391	C	[illegible]	[illegible]	[illegible]
	17	1.017	1.235	[illegible]	C	[illegible]	[illegible]	[illegible]
I-4	19	0.996	1.208	[illegible]	C	[illegible]	[illegible]	[illegible]
	20	0.996	1.201	[illegible]	C	[illegible]	[illegible]	[illegible]
	21	0.996	[illegible]	[illegible]	C	[illegible]	[illegible]	[illegible]
	22	[illegible]	1.205	[illegible]	C	[illegible]	[illegible]	[illegible]
	23	0.994	1.203	[illegible]	C	[illegible]	[illegible]	[illegible]
	24	0.994	1.194	0.344	C	[illegible]	[illegible]	[illegible]
I-5	25	1.018	1.073	0.0900	T	[illegible]	[illegible]	[illegible]
	26	1.019	1.070	0.0837	T	[illegible]	[illegible]	[illegible]
	27	1.019	1.074	[illegible]	T	[illegible]	[illegible]	[illegible]
	28	1.018	1.078	[illegible]	C	[illegible]	[illegible]	[illegible]
	29	1.020	1.076	0.0922	C	[illegible]	[illegible]	[illegible]
	30	1.020	1.077	[illegible]	C	[illegible]	[illegible]	[illegible]
I-6	31	1.018	1.107	0.149	T	[illegible]	[illegible]	[illegible]
	32	1.018	1.103	0.142	T	10840	[illegible]	[illegible]
	33	1.019	1.103	0.140	T	[illegible]	[illegible]	[illegible]
	34	1.020	1.104	0.140	C	11800	[illegible]	[illegible]
	35	1.019	1.102	0.147	C	10700	[illegible]	[illegible]
	36	1.019	1.102	0.138	C	[illegible]	[illegible]	[illegible]

[a] See Table 4.

TABLE 9. TORSION TEST RESULTS

Specimen							
Type*	No.	I.D. (in.)	O.D. (in.)	Area ($in.^2$)	Ult. Load (in. - lb.)	G (10^6 psi)	Ult. Stress (psi)
II-1	37	1.011	1.125	0.191	1730	1.5	17,793
	38	1.013	1.125	0.188	1700	1.7	17,749
	39	1.011	1.125	0.191	1680	1.7	17,279
II-2	40	0.996	1.108	0.185	2650	2.6	28,589
	41	0.995	1.115	0.199	2200	2.6	22,094
	42	0.996	1.114	0.195	2400	2.5	24,491
II-3	43	0.994	1.050	0.090	630	0.91	14,079
	44	0.993	1.050	0.091	590	0.90	12,972
	45	0.995	1.050	0.088	740	0.92	16,813
II-4	46	1.018	1.100	0.136	980	0.65	14,073
	47	1.017	1.102	0.141	990	0.77	13,718
	48	1.018	1.100	0.136	1080	0.73	15,508
II-5	49	1.002	1.134	0.221	2580	0.77	18,371
	50	1.001	1.132	0.219	2280	0.69	16,418
	51	1.000	1.131	0.219	2090	0.73	16,440
	52	1.006	1.139	0.224	2100	0.70	15,973
	53	1.006	1.129	0.206	2120	0.63	14,545
II-6	54	1.000	1.074	0.121	1110	1.65	23,078
	55	1.000	1.076	0.124	1020	1.56	20,603
	56	1.002	1.084	0.134	1110	1.54	18,921
	57	1.001	1.084	0.136	1090	1.56	18,490
	58	1.002	1.091	0.146	1070	1.62	20,300

*See Table 5.

TABLE 10. BEARING TEST RESULTS

Specimen		Thickness (in.)		Pin Dia.	Ultimate Load (lbs)	Ult Bearing Stress (psi)
Type*	No.	Steel	Comp.			
	59	0.10	0.258	3/16	2,475	51,000
III-1	60	0.10	0.260	3/16	2,535	51,800
	61	0.10	0.258	3/16	2,500	51,500
	62	0.10	0.257	1/4	3,910	60,900
III-2	63	0.10	0.257	1/4	3,625	56,400
	64	0.10	0.257	1/4	3,825	59,500

*See Figures 26 and 27.

BALLISTIC VULNERABILITY EVALUATION (HHI-SPONSORED TEST)

A ballistic vulnerability test was undertaken to determine the vulnerability of the CTS to the 23mm HEI-T projectile threat. The tests on two simulated tailboom shells determined the suitability of design refinements to be incorporated in the CTS. The test articles consisted of two tapered cylindrical specimens designed to simulate the CTS structural shell from Sta 450 to 547. Both specimens were of sandwich construction with adapters used to interface with the metal tailboom ballistic test fixture. The specimens were honeycomb sandwich construction with composite face sheets and Nomex core. Figure 31 shows the details of construction. Specimen #1 had 0° graphite/±60° E-glass/90° Kevlar skins with 200 x 200 aluminum wire mesh on the outside surface for lightning protection. Specimen #2 had 90°/±12.5°/90° all graphite skins with no lightning protection. Instrumentation consisted of a set of photoelectric velocity screens and electronic time interval counters for determining the impact velocity of the projectiles, and a load cell to verify load conditions. Both high speed and still photographic records were made.

An equivalent hover load was applied to the loading fixture (approximately 1,500 lbs) during and after ballistic impact. Each test specimen was subjected to one Soviet 23mm HEI-T projectile with an MG-25 delay fuze at 1,500 ±50 ft per sec in an aligned attitude at Station 510. Figure 32 shows the test setup and impact location.

Specimen #1 shows less severe damage than Specimen #2 due to the excellent energy absorption capacity of the Kevlar and E-glass, although both specimens are considered successful in defeating the 23mm threat.

Ballistic damage to Specimen #1 consists of a clean entrance hole and three zones of exit damage. See Figure 33 through 36. Zone 1 consists of an area of approximately 8 inches in diameter on the back side of the specimen on the shot line. Damage in this area consists of approximately five holes ranging in size from one to two inches in diameter. One hole is accompanied by a torn face sheet approximately 2 by 6 inches. Zone 2 consists of an area approximately 24 inches in diameter on the back side of the specimen. Damage in this area consists of a large number of small perforations (less than one inch in diameter) spaced relatively widely. Zone 3 consists of two areas approximately 30 by 6 inches along the top and bottom of the specimen just forward of the center line. Damage in this area is most severe with a multitude of small (less than one inch) holes accompanied by some local tearing and delamination of the outer face sheet. Delaminations and tears range up to approximately 3 by 4 inches, maximum. No large holes or delaminations appear in the specimen and sufficient cross section remains intact to assure safe flight.

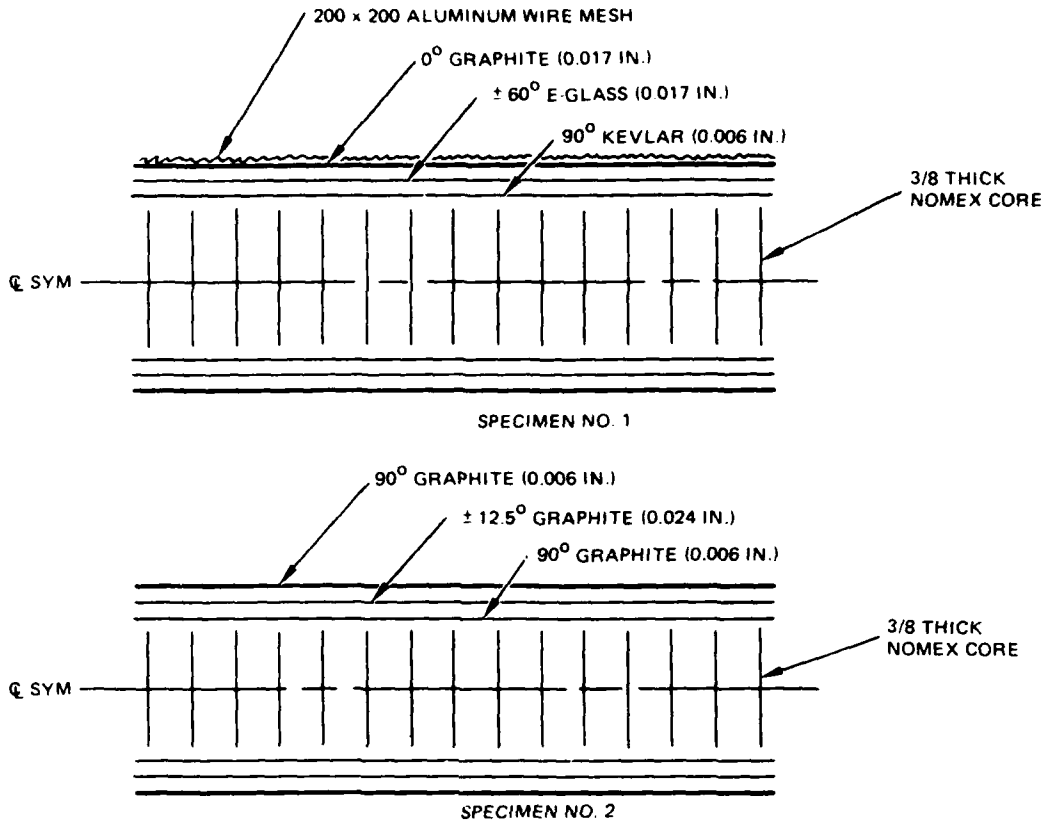


Figure 31. Ballistic Test Specimen Construction

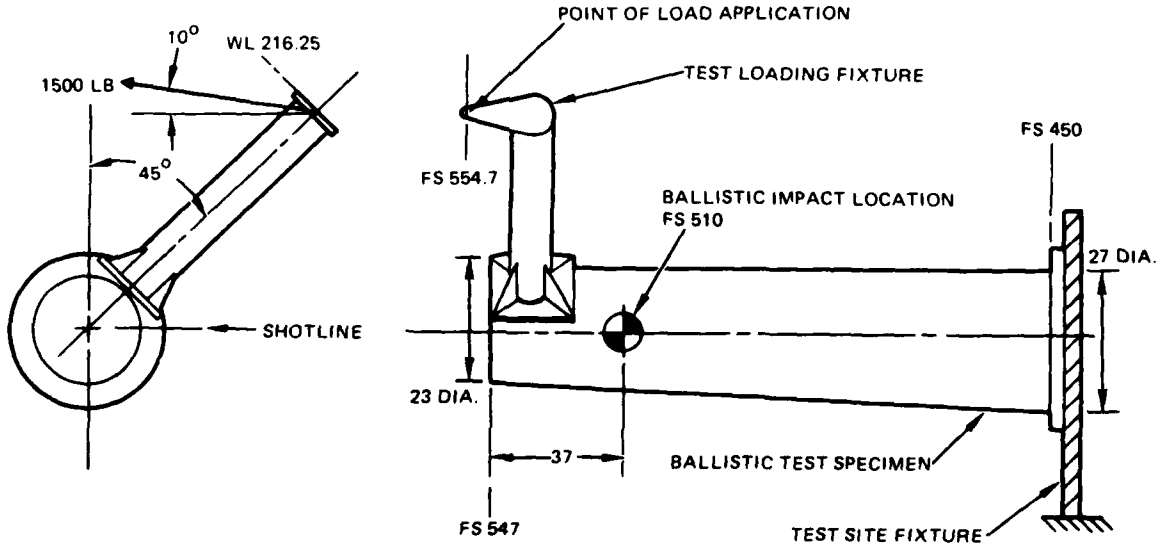


Figure 32. Ballistic Test Setup and Impact Location

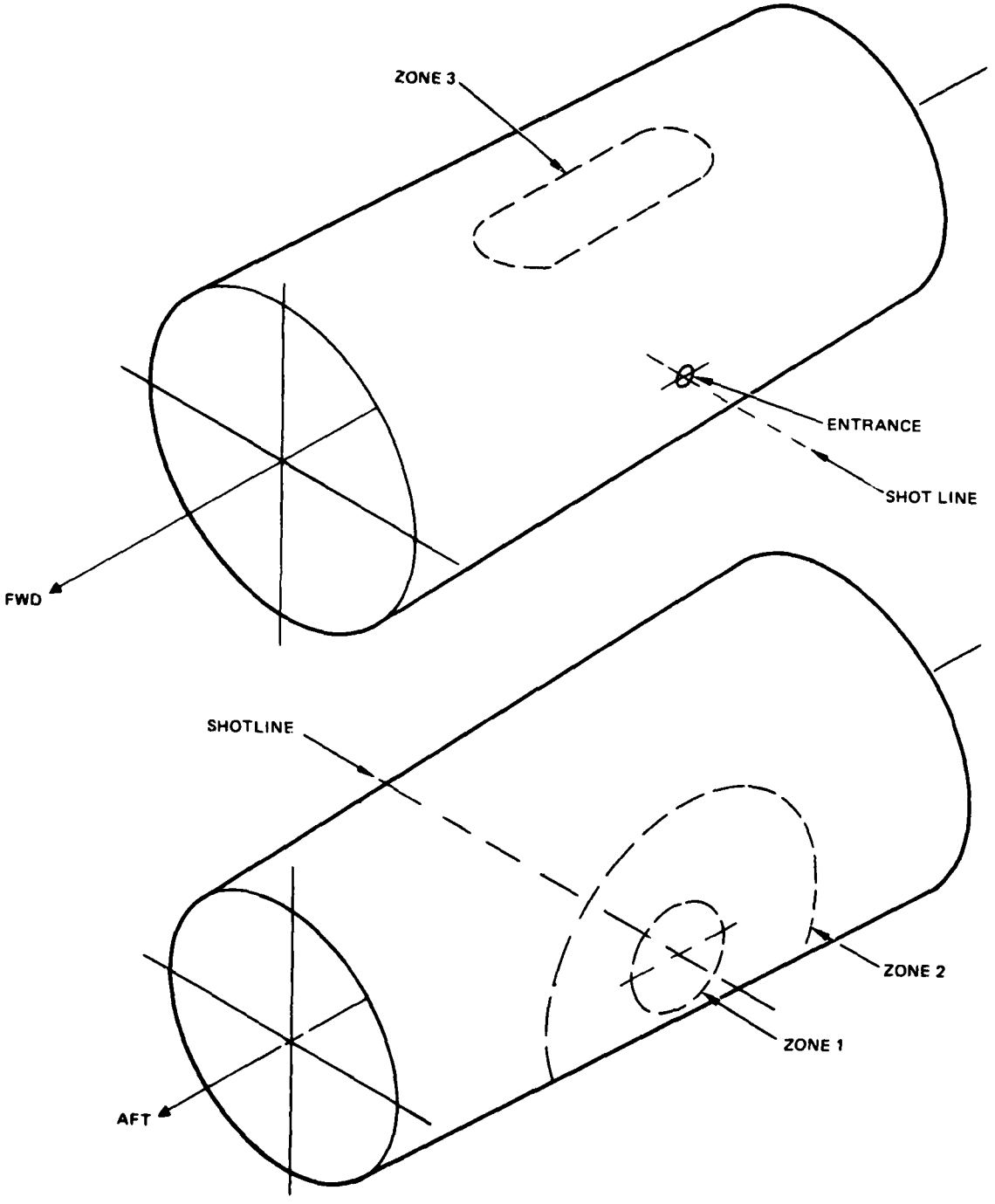


Figure 33. Ballistic Damage Zones.

Figure [illegible] [illegible]

[illegible] 10882-7

Figure [illegible] [illegible] Drainage

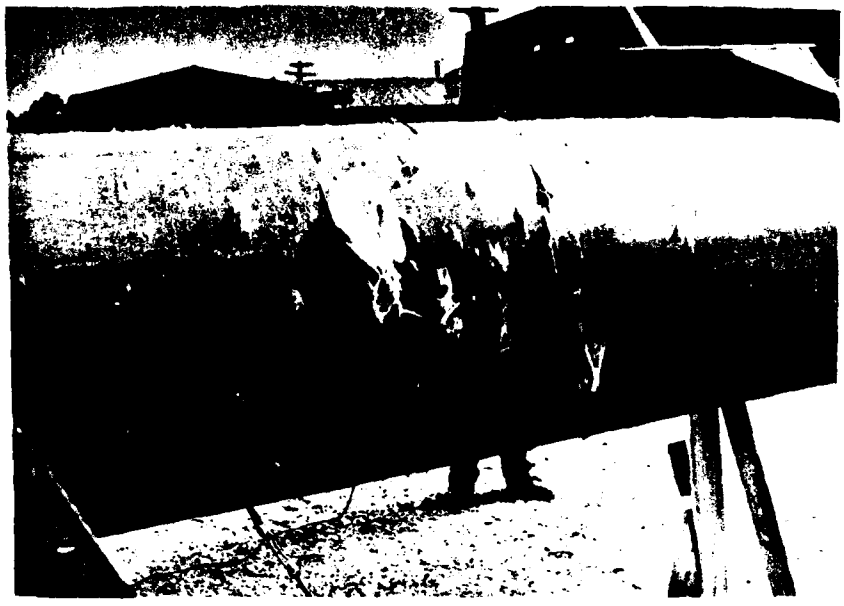


Figure 36. Specimen #1 -- Rear View With Damage

Damage to Specimen #2 is more severe than to Specimen #1. However, residual strength of the specimen is not reduced sufficiently to prevent carrying the hover load. See Figures 37 through 40. Referring to the same zone designations as for Specimen #1, Zone 1 damage is quite similar with slightly more delaminations of the skin. Zone 2 damage is also very similar to that for Specimen #1, although the perforations through the all-graphite skin are much cleaner due to the brittleness of the graphite fibers compared to the Kevlar and fiberglass fibers. Several of the holes are accompanied by minor local delaminations of the skin. Zone 3 damage to Specimen #2 is much more severe than Specimen #1. Although the majority of the holes are smaller, several large holes are present with an area of approximately 6 by 18 inches torn away from the honeycomb core on the lower side.

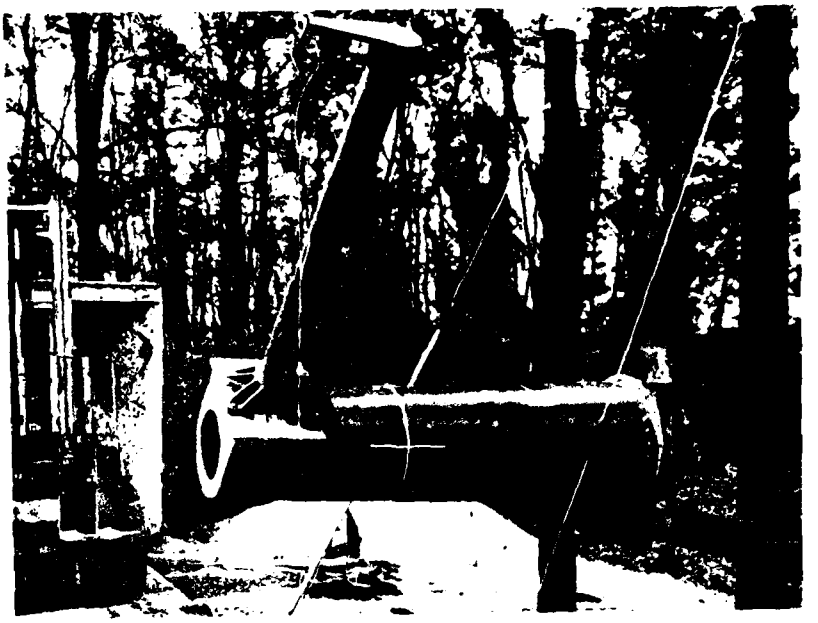


Photo 80-10881-13

Figure 57. Ballistic Specimen #2 -- Undamaged

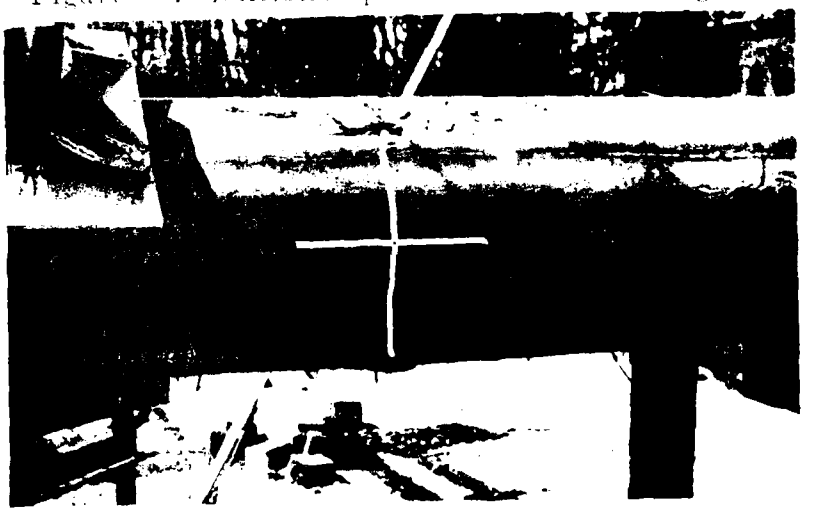


Photo 80-10881-22

Figure 58. Specimen #2 -- Front View With Damage

Photo 80-10881-25

Figure 39. Specimen #2 -- Close-up Front View With Damage

Photo 80-10881-23

Figure 40. Specimen #2 -- Rear View With Damage

LIGHTNING PROTECTION EVALUATION (HHI-SPONSORED TEST)

Preliminary lightning tests were conducted on two graphite/Kevlar 12- x 24-inch flat test panels that were representative of the tailboom laminated sandwich in the Sta 370 region (see stacking sequence in Table 2 of Figure 9(The tests consisted of high current tests in which 200 kiloamperes (or as much as the specimen would pass) was fired through the sample from end to end, and a 100 kiloampere discharge fired as a restrike into an inboard area of the specimens.

The test arrangement is shown in Figures 41 and 42. The test results are summarized as follows:

- A 150,000 ampere end-to-end discharge on these flat panels causes only very slight damage, but the current is limited by the resistanc of the graphite. An additional discharge across half the panel (to lower the resistance) also causes only slight damage. See Figures and 44.

- A 100,000 ampere discharge inboard on the sections representing a swept stroke causes delamination of the outer skin of both test samples over an area approximately ten inches square. The damag appears to be severe at first, but it is determined not be flight critical. See Figures 45 and 46.

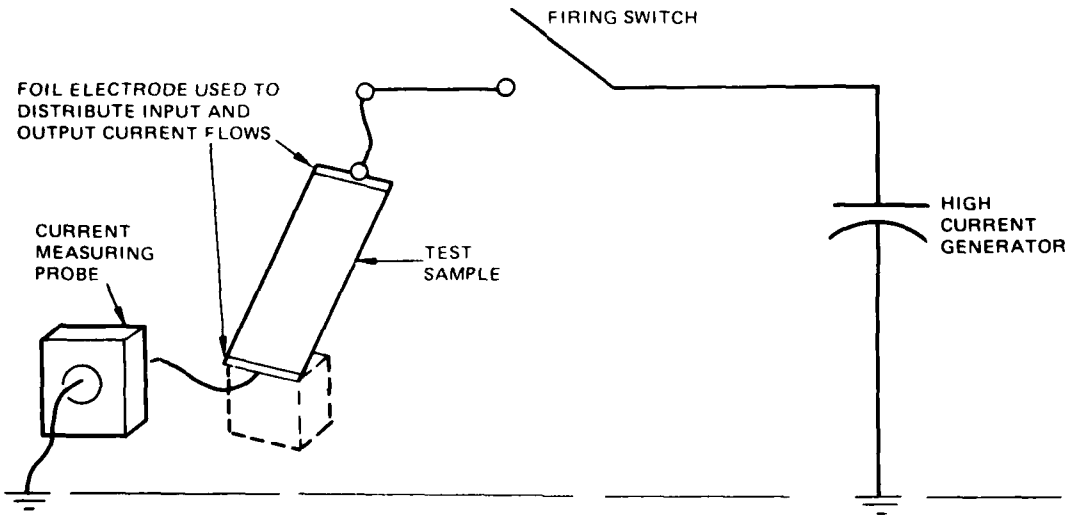


Figure 41. Test Arrangement for Stationary 200,000 Ampere Discharge)

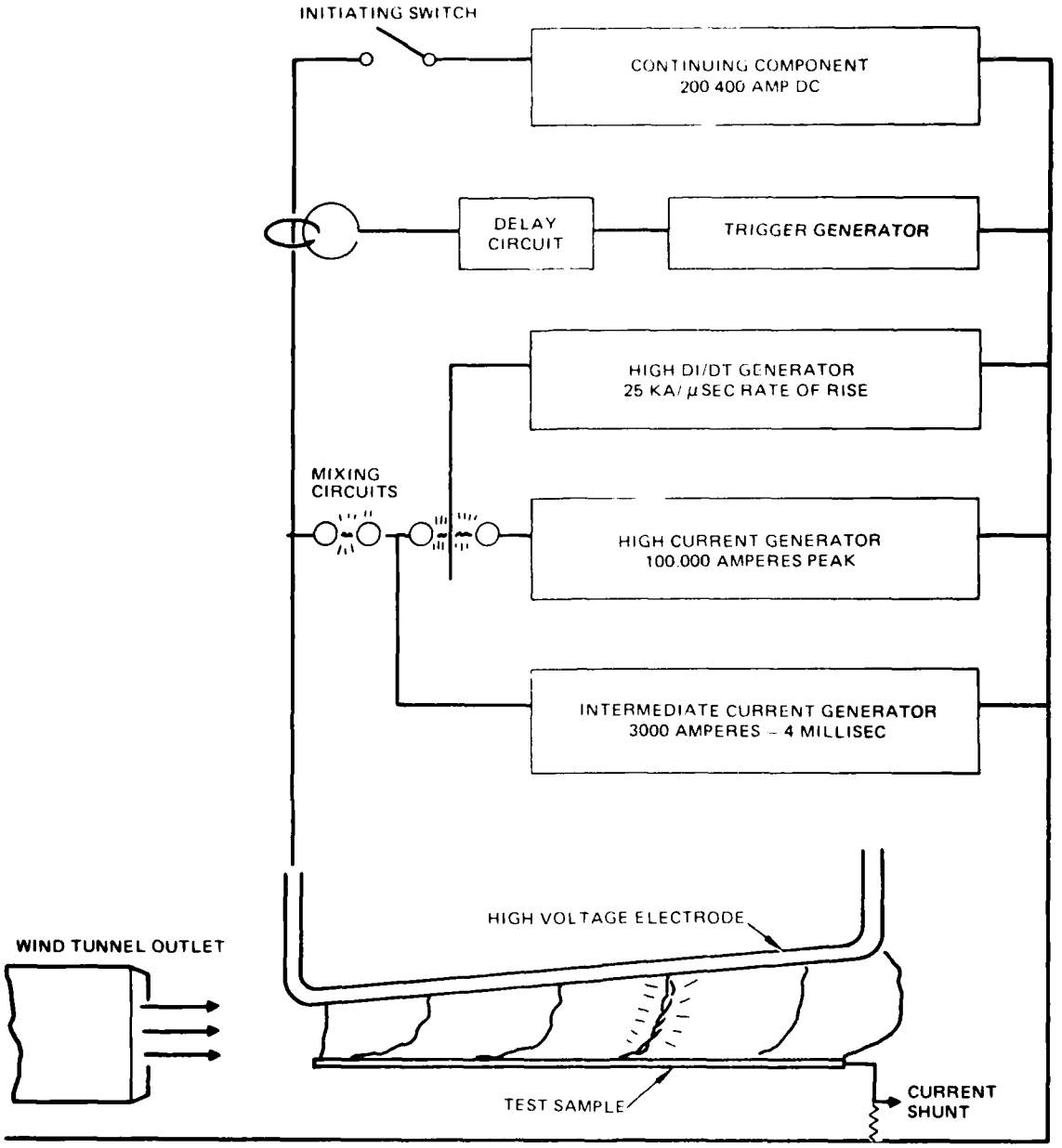


Figure 42. Test Arrangement for Swept Stroke Tests

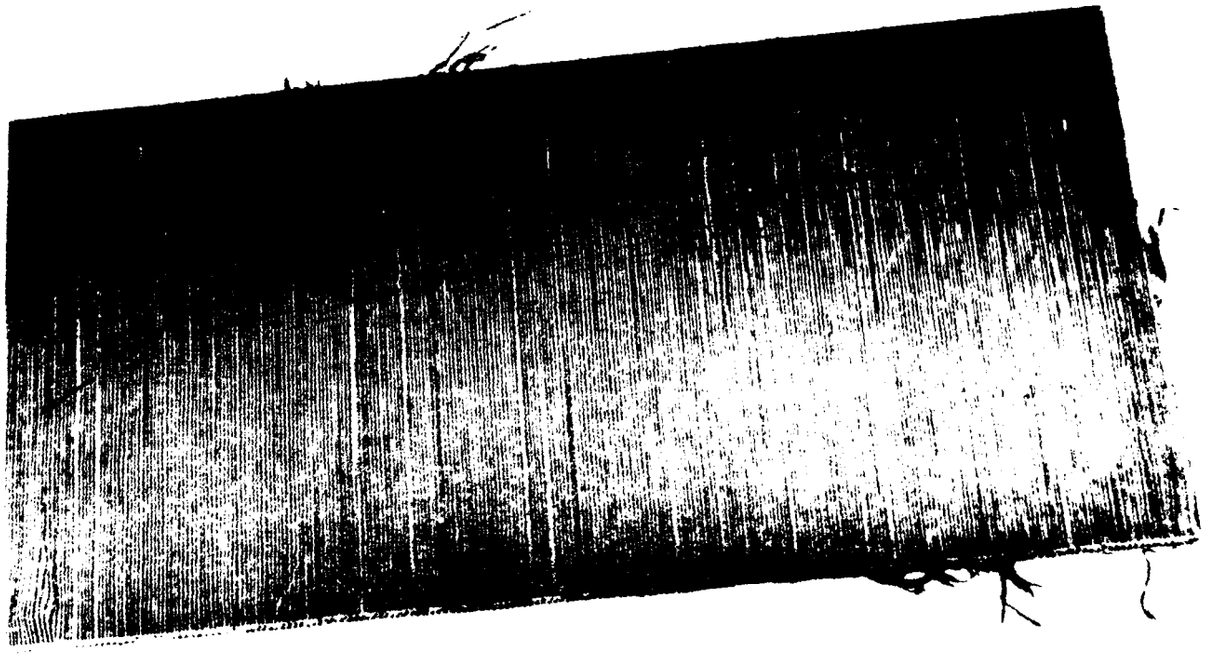


Photo No. 4740

Figure 43. Stationary Discharge Damage -- Specimen 1

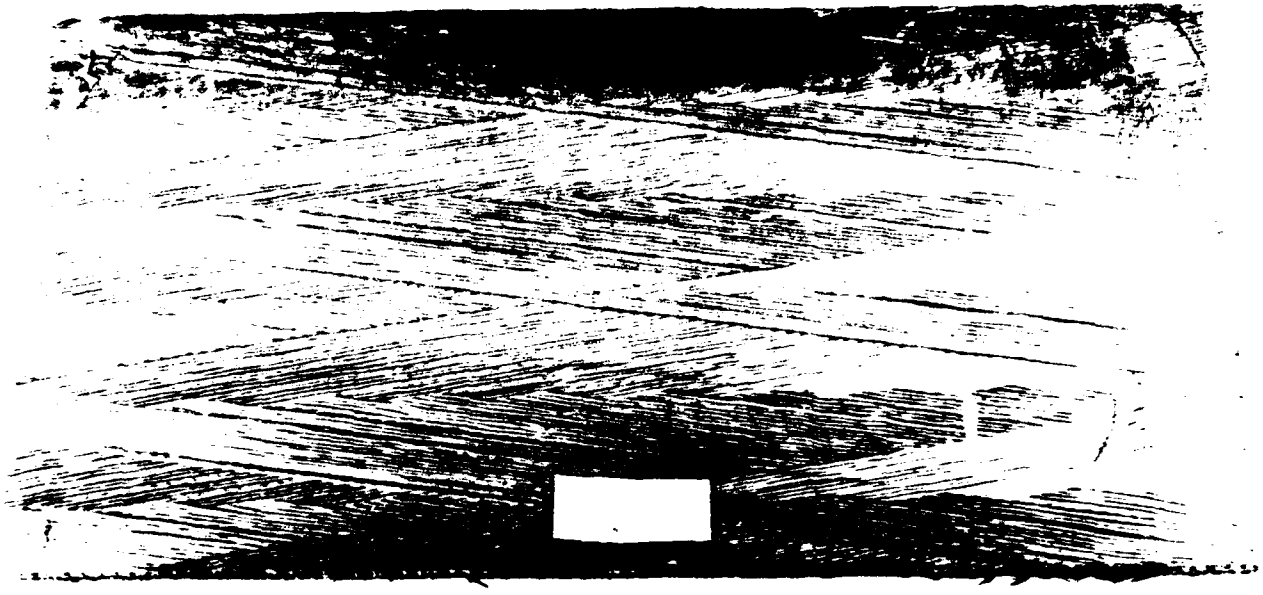


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Figure 44. Stationary Discharge Damage -- Specimen 2

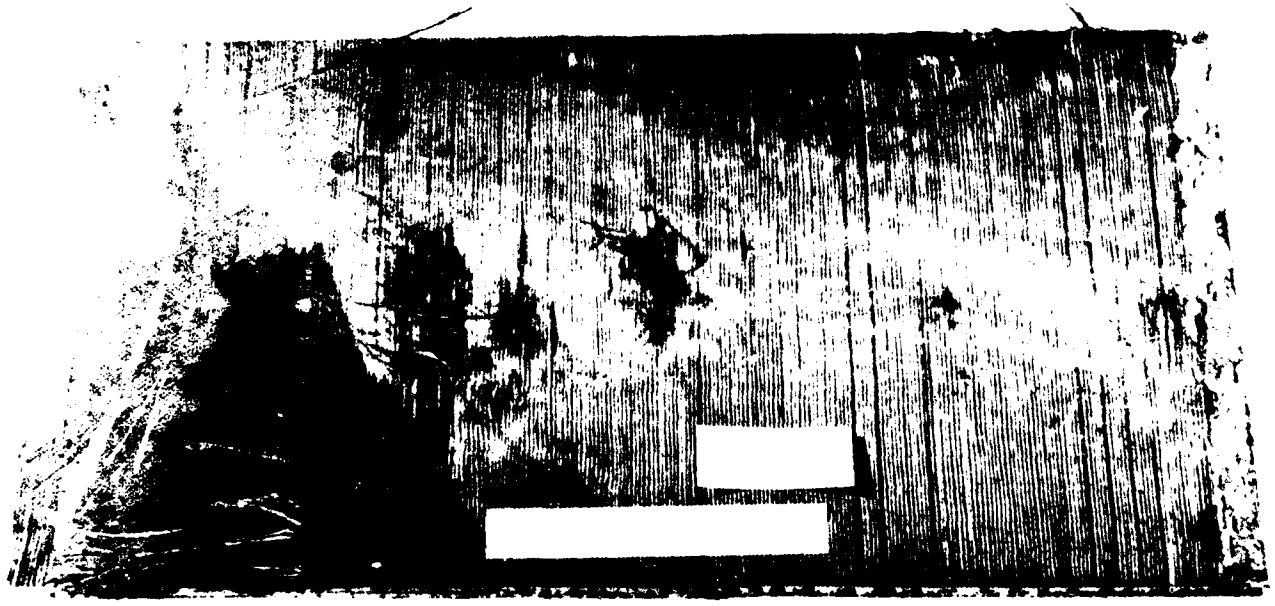
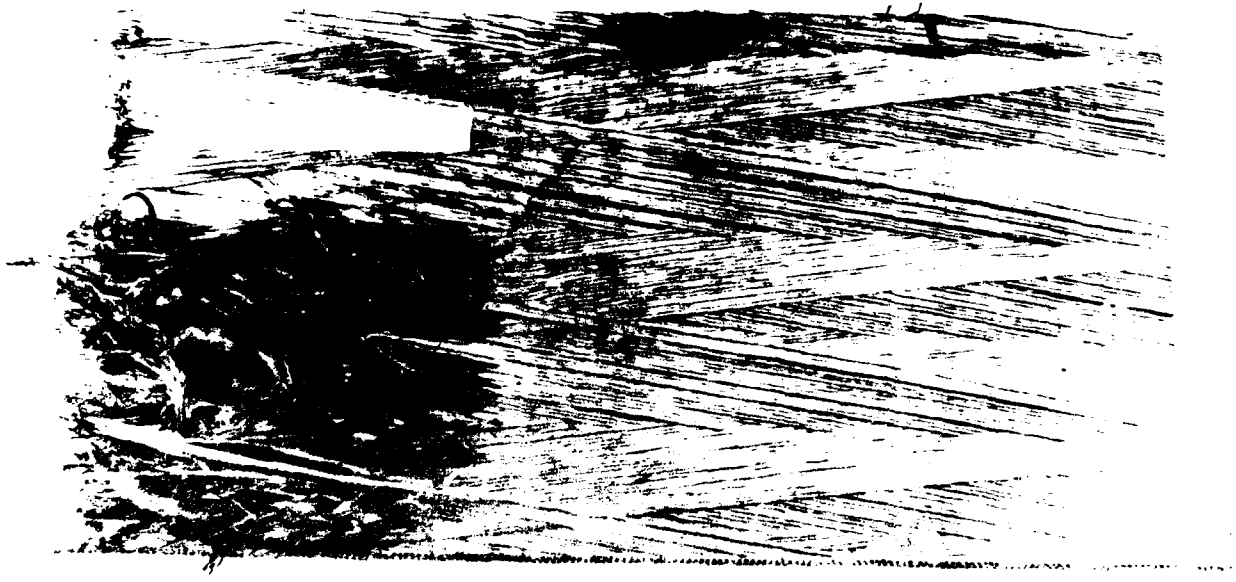


Photo 80-4741

Figure 45. Swept Stroke Discharge Damage -- Specimen 3



The tests may be summarized as having caused little damage if the test of 150,000 amperes from end-to-end of the test section. However, 200,000 ampere discharges cause extensive delamination of the outer skin if a direct contact with the arc. The damage in the latter case is less severe than the ballistic damage caused by a 23mm HEI-T projectile, and this is considered to be non-flight critical. In either case, the damage is considered repairable, even in the event of a swept lightning strike directly to the tailboom.

It is concluded that because the skin can withstand a swept lightning stroke (which is noted as an unlikely event) that further protection need not be provided.

ANTENNA INTEGRATION EVALUATION

A feasibility study was conducted for the #1 FM antenna that is integrated into the trailing edge of the vertical tail to determine if its operation is compatible with the composite tail section. The study included testing of the various composite materials proposed for use, and included comparative impedance, patterns, and gain measurements in the FM frequency band for both aluminum and graphite/epoxy composite tail section mockups. Antenna impedance, pattern, and gain were measured. The purpose of the study was to determine if the existing FM #1 Vertical Stabilizer Trailing Edge Antenna designed for installation on a metal tail section would provide acceptable performance when installed on a composite tail section, and to provide recommendations for antenna redesign if so indicated by the study.

The electrical properties of materials were studied to determine the construction of the composite mockup to simulate the electrical characteristics of the tail structure closely and with reasonable cost. Unidirectional graphite fabric was used in an epoxy resin wet layup to electrically simulate the semiconducting members of the mockup. Since structural strength was not needed for the mockup, part thicknesses were chosen to simulate the R.F. properties of the structure. Graphite parts were restricted to 0.06 inch thickness, thereby limiting the cost of the mockup and minimizing fabrication time. This amounts to six plies of the unidirectional graphite fabric use. (90% unidirectional graphite fabric, T-300, 3000 filament count with 10% glass fill fibers. Woven Structures Style No. ES 1[illegible].)

One composite material designated for study and test was the Thorstrand (aluminized fiberglass)/Kevlar/graphite/epoxy laminate. The electrical properties of this candidate are very difficult to analyze and test because the Kevlar dielectric layer separates the two semiconducting layers. Resistance measurements were made on the Thorstrand alone and on graphite epoxy alone. Manufacturer's resistivity data was also obtained. Composite materials providing improved strength-to-weight characteristics over aluminum fall in the category of electrically non-conducting (dielectric) or semi-conducting materials, whereas aluminum is highly conductive to direct electrical current (DC) and is known as a good conductor. Based on these materials, electrical tests, and the lightning tests, it was determined that the Thorstrand was not needed for the mockup, and only graphite and fiberglass were selected for its construction.

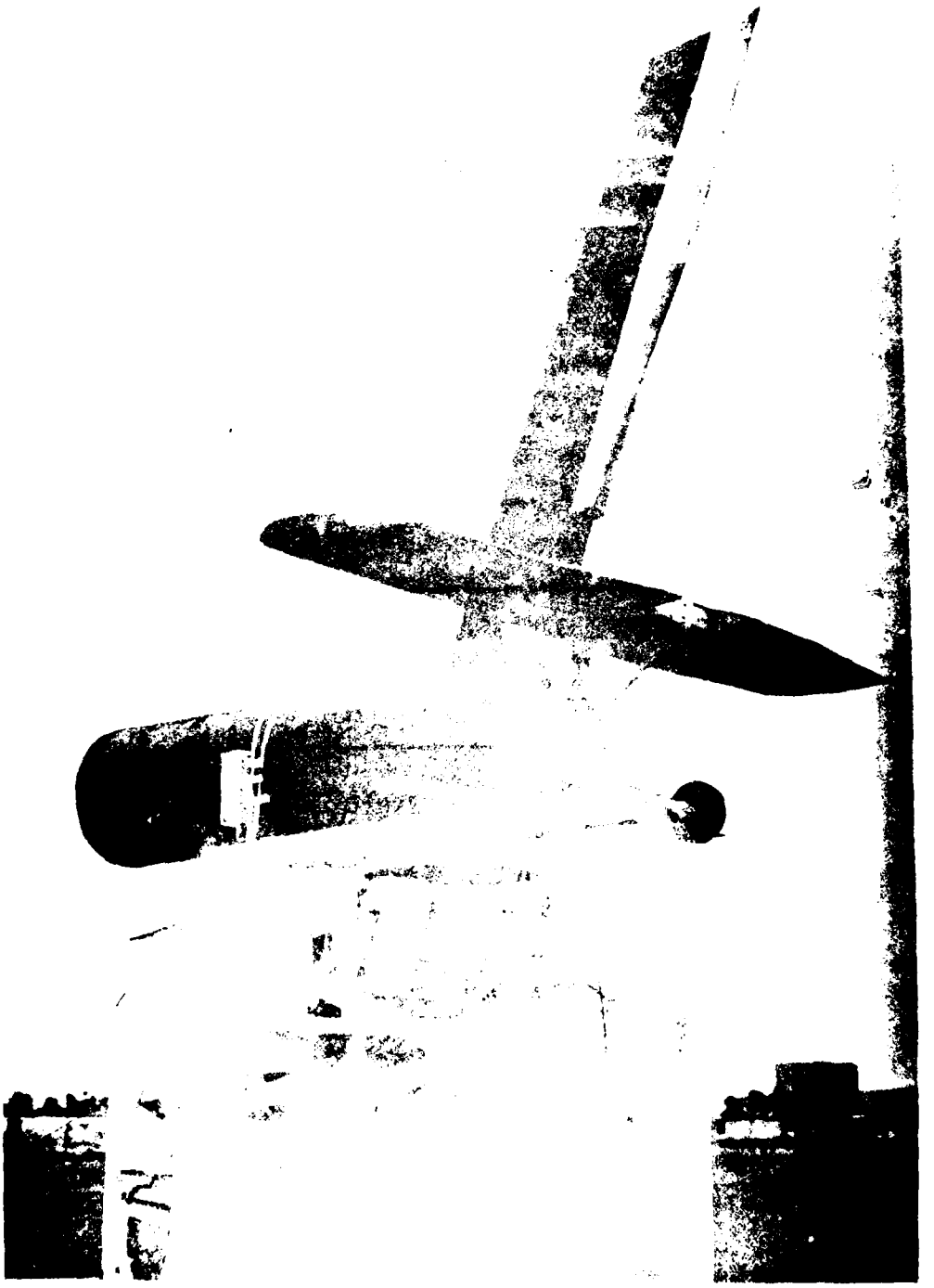
For antennas in the VHF frequency region and below, the airframe on which they are installed form a part of the antenna and contribute to its impedance, radiation pattern, and gain characteristics. The degree to which this is true depends upon the antenna configuration as well as the airframe configuration and the physical relation between antenna and airframe. If the airframe is highly conductive to the RF energy, these characteristics can vary widely when compared to a nonconducting or semi-conducting airframe.

To satisfy the requirements of this study, it was necessary to have two full scale AH-64A tail mockups. Since the trailing edge antenna is designed to operate on the metal version of the tail section, a metal mockup from the AH-64A program was provided for this purpose. All parts of the tail structure from Sta 450 aft that are necessary to electrically simulate the AH-64A metal tail section are included. The composite tail section mockup includes the tailboom skin, vertical spar box skin, and the ribs and spars within the stabilator. Stacking angles and the number of plies are maintained as closely as possible. The stabilator skin is fabricated of epoxy fiberglass. Additional parts of the tail section included in the mockup are tail rotor shaft and gearbox, vertical rotor driveshaft, and tail landing gear. The stabilator is movable in elevation within the +10° to -45° limits. Figure 47 shows the AH-64A graphite tail section mockup installed on the tower/positioner. Figure 48 is a schematic of the equipment.

The Vertical Tail Trailing Edge (VSTE) Antenna (P/N 7-211122630) is a *folded dipole antenna* integrated into the trailing edge fairing of the AH-64A helicopter vertical tail. An aluminum sheet metal spar at the forward side of the antenna and forms the "folded" portion of the antenna element.

Swept frequency impedance measurements were performed in the 30 to 70 MHz frequency range of the trailing edge antenna installed on both the aluminum and composite tail section mockups. These measurements enable a judgement to be made as to the comparative level of difficulty to be anticipated in impedance matching the antenna to the 50-ohm transmission line in the two different installation environments. A network analyzer was used to measure the antenna impedance.

Figures 49 and 50 show the impedance plots on impedance coordinates of the Smith Chart, with Several Frequency points in the FM band marked. These impedance plots show that the antenna impedance variation between the two installations is small. As a result, major impedance matching network compensation is not required to provide antenna impedance performance compatibility on the metal and composite tail sections. At most, some circuit component value changes in the matching network may be required. Antenna impedance is not appreciably affected by changing stabilator position on the metal mockup.



Mockup

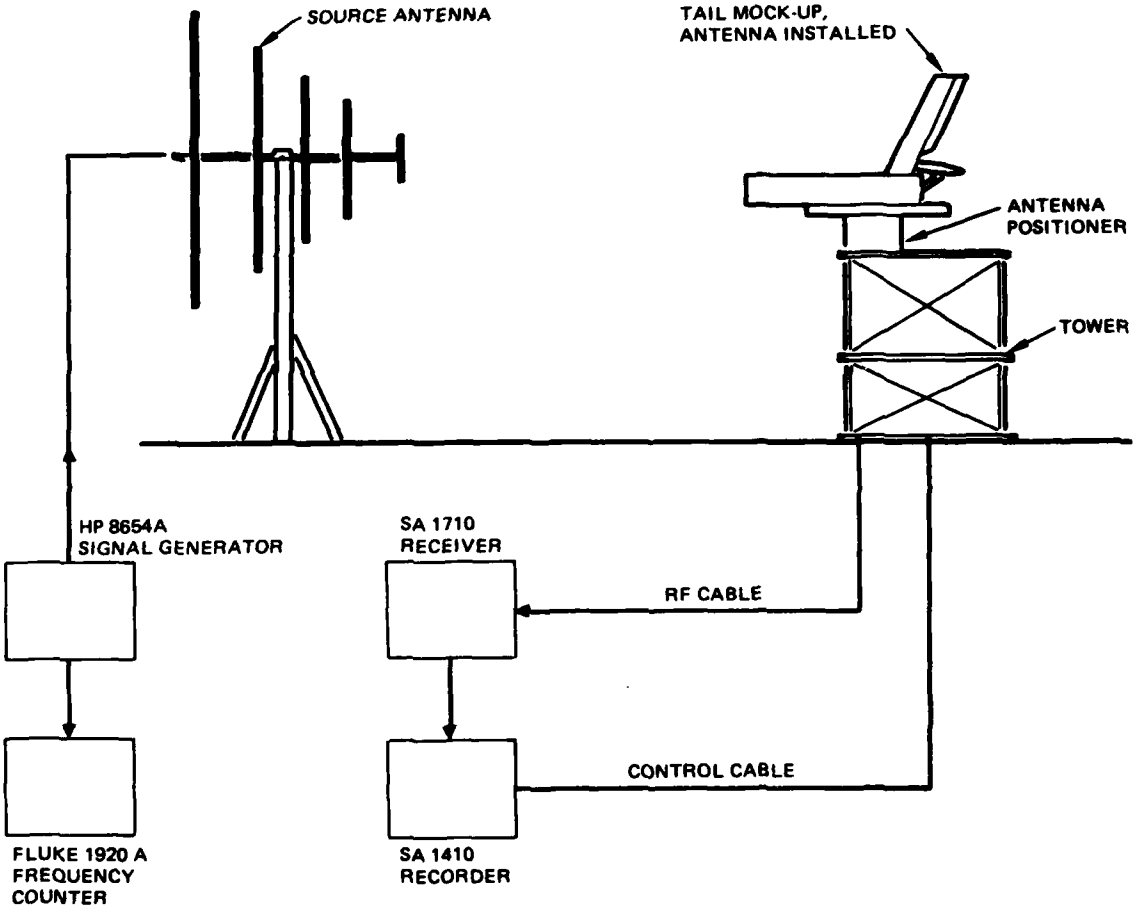


Figure 48. Radiation Pattern Test Setup.

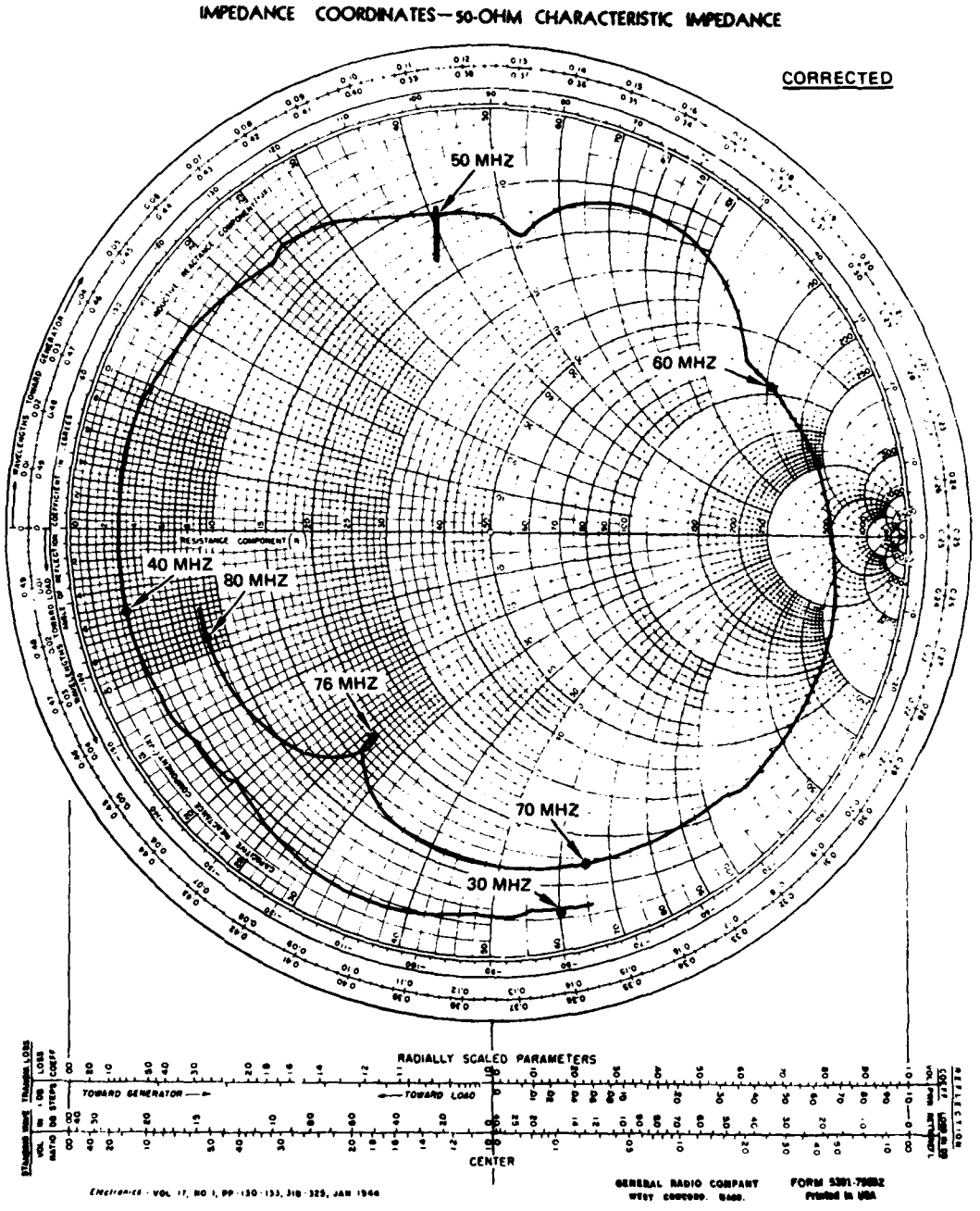


Figure 49. AH-64A Trailing Edge Antenna on Aluminum Mockup

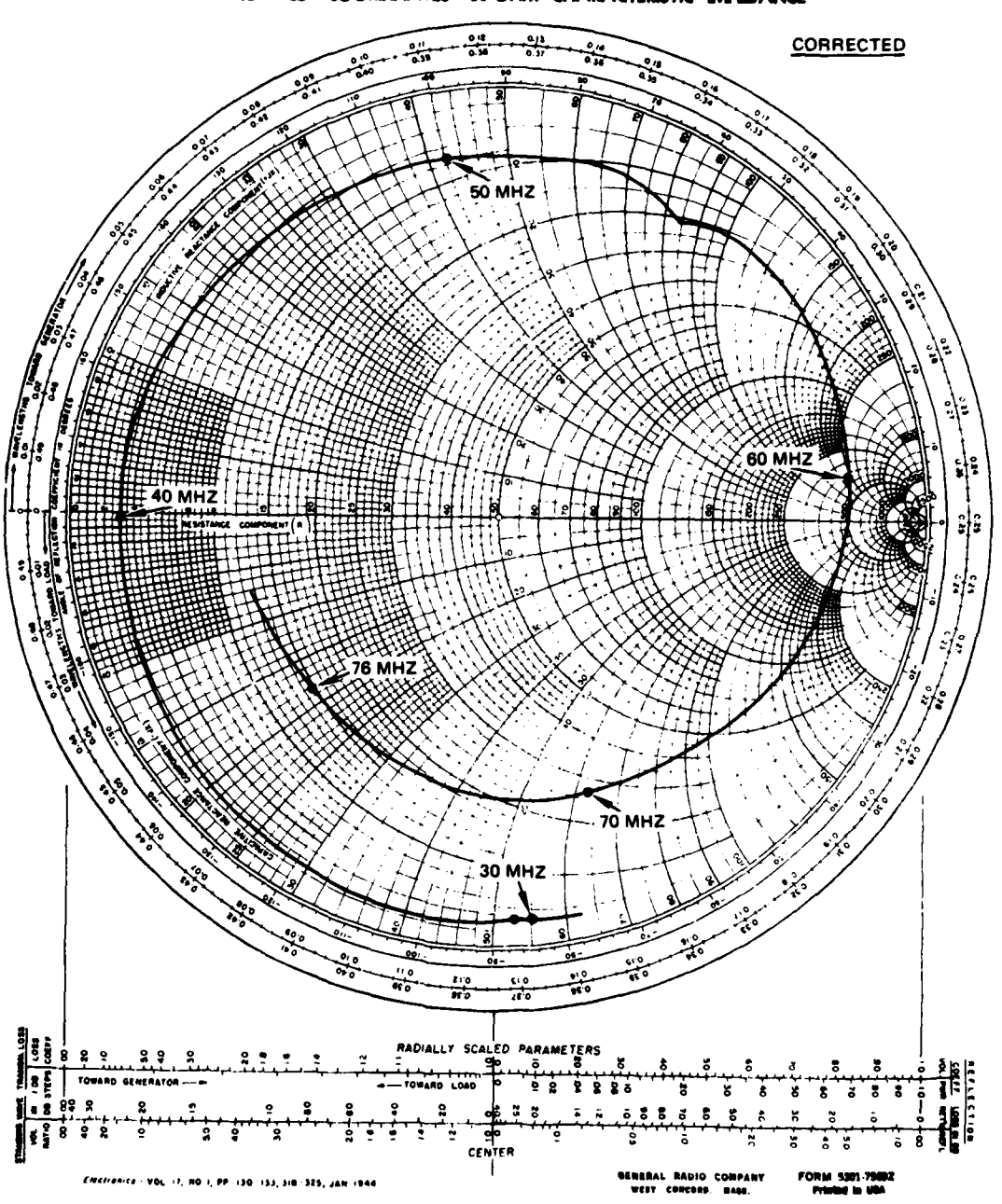


Figure 50. AH-64A Trailing Edge Antenna on Graphite Mockup.

Azimuth radiation pattern and gain measurements were performed at six frequencies in the VHF-FM band on the trailing edge antenna installed on both the aluminum and graphite tail mockups. The test setup is shown in Figure 48. The radiation patterns are shown in Figures 49 and 50. Each pattern contains a gain reference to allow the relative gain of the antenna on the graphite mockup to be compared to that of the antenna on the aluminum mockup. No attempt has been made to measure the absolute antenna gain for this program, since relative gain provides the necessary data for comparison. As shown by the patterns, the major difference is found in the 50 MHz pattern for the graphite mockup. This pattern contains a deep azimuth null and exhibits much lower gain than that for the metal mockup pattern. This condition is not characteristic of this antenna. The deep null can be caused by a parasitic element located on the mockup forward of the antenna. Since the only part of the mockup that has this characteristic is the tail rotor vertical driveshaft, it was removed and the patterns were rerun at one MHz intervals from 40 to 60 MHz. The null remained at frequencies of 49, 50, and 51 MHz, indicating that it is not caused by the driveshaft. This pattern and gain deficiency is believed to be caused by the test environment.

The test tower contains metal positioner, supporting structure, and connecting cables that can act as parasitic elements and are resonant at one or more frequencies. Resulting radiation patterns can contain both azimuth and elevation nulls. The azimuth null is evident in the azimuth pattern. An elevation null located on or near the horizon can also result in the lowered gain. But, as shown by the 50 MHz pattern for the aluminum mockup, a null is not present. The metal skin of the mockup is responsible for this change. The conducting skin has most likely upset the parasitic arraying effect of the test structure to eliminate the null entirely or shift it to a frequency not tested when the antenna is installed on the aluminum mockup. A less severe pattern deficiency occurs at 60 MHz on the metal model. The two deeper side nulls and lower relative gain are caused by the parasitic arraying effect of the antenna tower structure and the aluminum mockup.

Antenna pattern tests were run on the metal mockup with stabilator up 10 degrees and down 45 degrees for comparison with the 0 degree position. Negligible differences were noted at the six test frequencies. These tests were not repeated for the composite mockup because the less conductive stabilator (graphite spar/rib -- Kevlar skin) is anticipated to cause an even smaller effect.

Antenna impedance data for both mockup installations shows only minor additional impedance variation between the two mockups. Impedance tests also show that antenna impedance is not appreciably affected by changing stabilator position. Hence, major impedance matching compensation is not required.

The comparative gain performance of the trailing edge antenna installed on the two mockups is quite favorable as Figures 51 through 56 show. For four out of the six test frequencies, the gain difference is 2 dB maximum. At the two remaining test frequencies, the gain varies 6 to 8 dB due to the effects of the test environment. It was concluded that this trailing edge antenna installed in the graphite/epoxy tail section meets the Contract requirements for the #1 VHF-FM antenna.

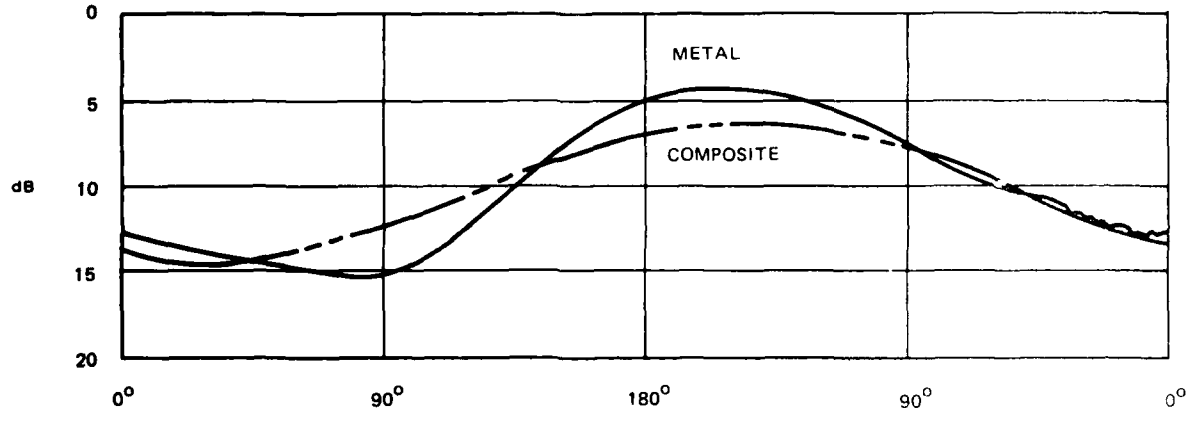


Figure 51. Antenna Pattern -- 30 MHz

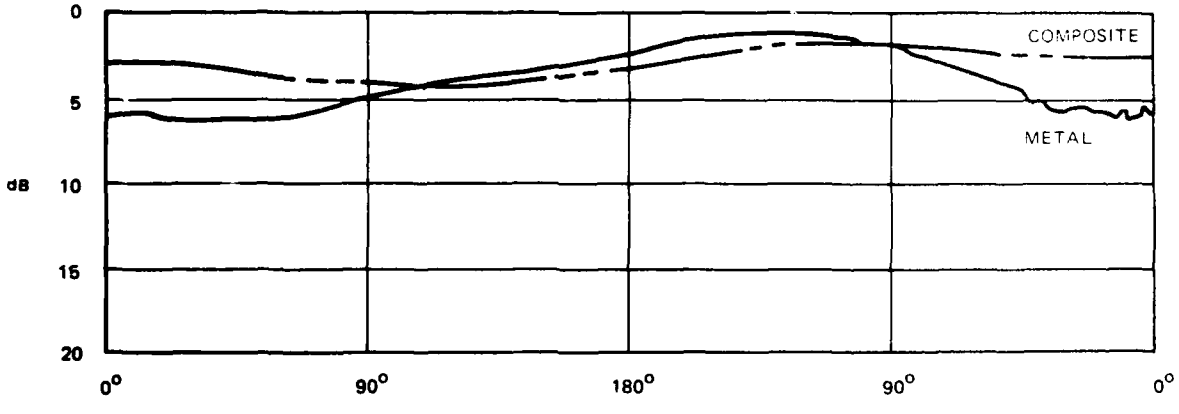


Figure 52. Antenna Pattern -- 40 MHz

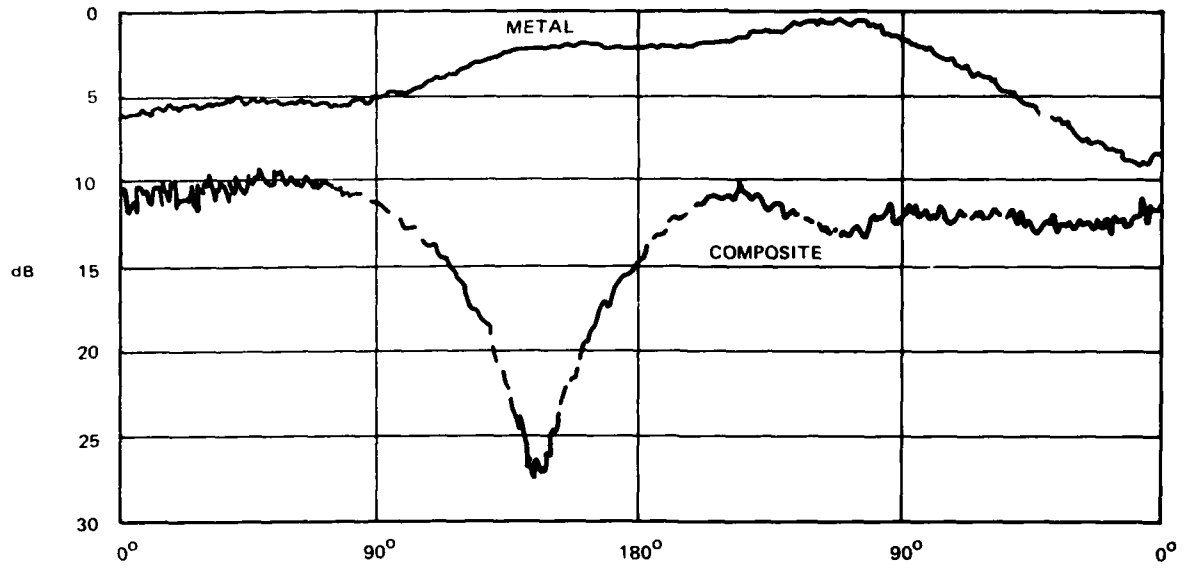


Figure 53. Antenna Pattern -- 50 MHz

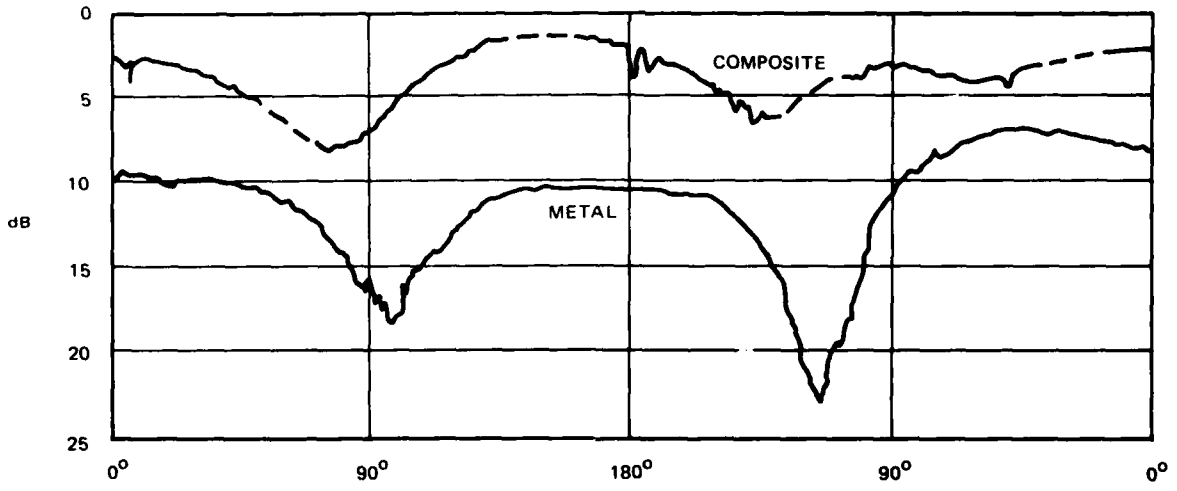


Figure 54. Antenna Pattern -- 60 MHz

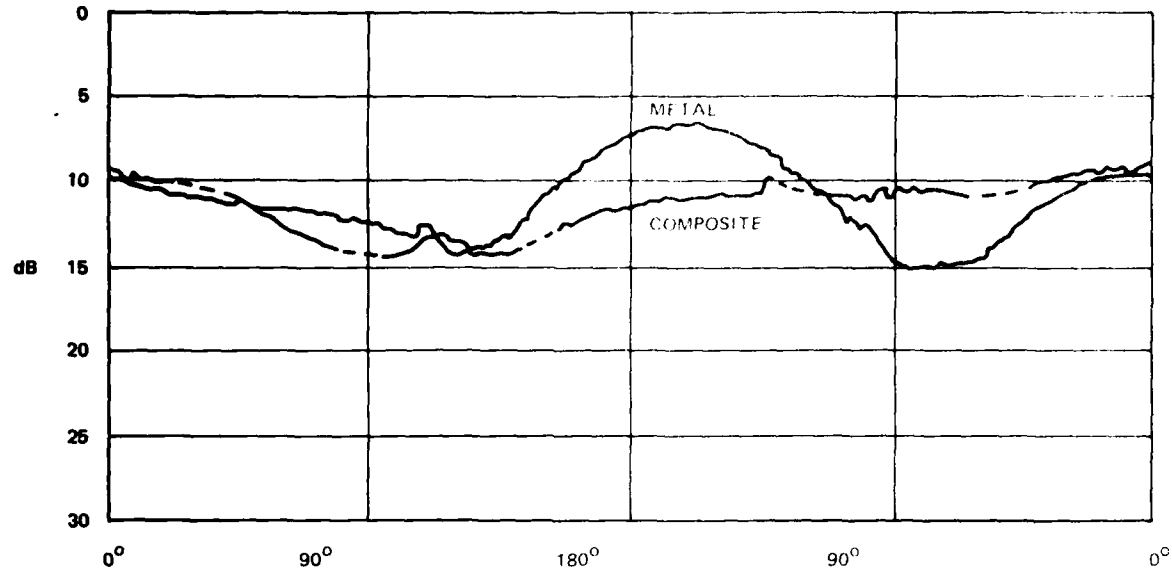


Figure 55. Antenna Pattern -- 70 MHz

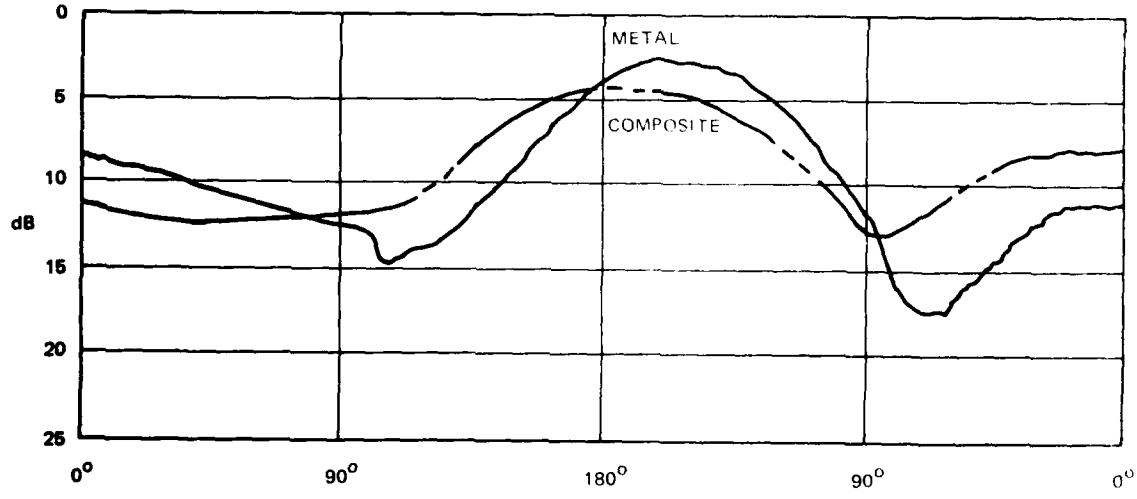


Figure 56. Antenna Pattern -- 76 MHz

MANUFACTURING TECHNOLOGY REFINEMENT

WET FILAMENT WINDING METHODOLOGY

The previous work that HHI and FSI had accomplished in the field of advanced composites primary structures was primarily in the field of the wet filament winding (WFW) cocure process. The CTS MM&T program was structured to continue the use of this process to the greatest extent possible, and refine it for the special requirements of the three CTS components.

The WFW process is summarized in Figure 57. This figure shows how bands of rovings are laid onto a rotating mandrel along geodesic paths until the entire surface of the mandrel is covered with a layer of filaments oriented in a ± angle pattern. This process can lay filaments onto a mandrel at any angle between approximately ±5 degrees and 90 degrees relative to the axis about which the mandrel rotates. Because the roving band follows a geodesic path to avoid slipping, the angular orientation is constant ± angle from end to end on a cylindrical mandrel. But it is a variable from end to end if the mandrel is conical (small ± angle at the large end to large ± angle at the small end). Each end of the mandrel has a suitably shaped dome for securing the filaments and guiding them smoothly as they turn around to make the reverse pass along the mandrel. It has been demonstrated that the mandrel need not have a circular cross section – rectangular and triangular cross sections are readily wound.

The band of wet rovings is created by passing a number of dry filament rovings (approximately 16 in number) through a resin impregnator such as that shown in Figure 58. It meters resin into the filaments to achieve a fixed fiber volume ratio. All of the filaments may be alike, such as graphite, or they may alternate, for example graphite and Kevlar, to form a hybrid composite. A dispensing eye whose travel along the mandrel is geared to the rotation of the mandrel guides the band of wet filaments onto the mandrel at the desired angle.

The tailboom and the vertical tail spar of the CTS are approximately cylindrical in shape and so are prime candidates for filament winding. The stabilator skin is also appropriate for WFW since its stacking sequence is 90/±45/90 degrees (90 degrees being the chordwise direction) and can be wound on a cylindrical mandrel, cut off, and formed to shape over an airfoil-shaped mandrel. With the basic concept of WFW being well established, the primary manufacturing development work for the CTS was the refinement of the details of how to wind each component.

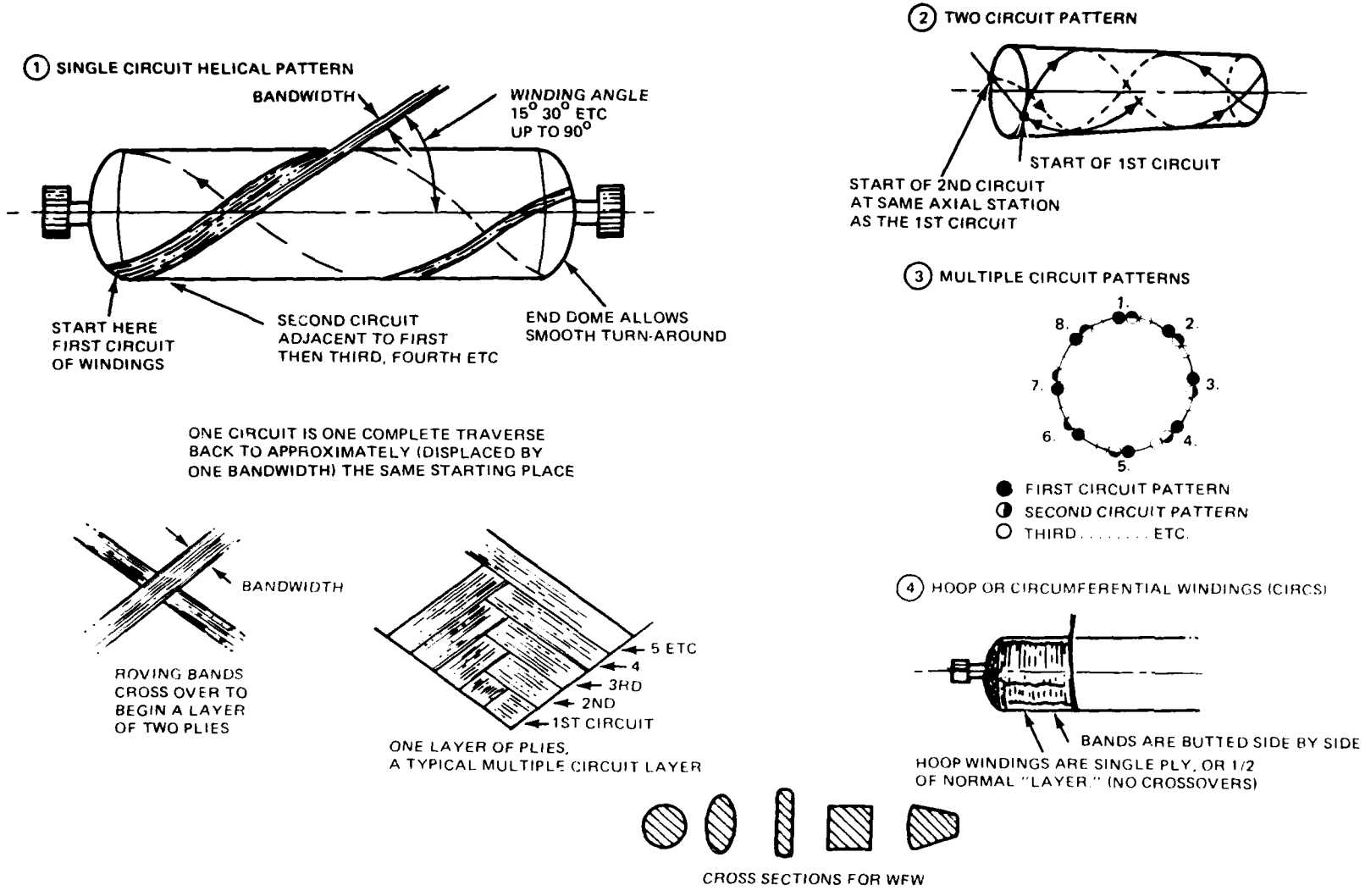


Figure 57. Tubular Element Filament Winding Process

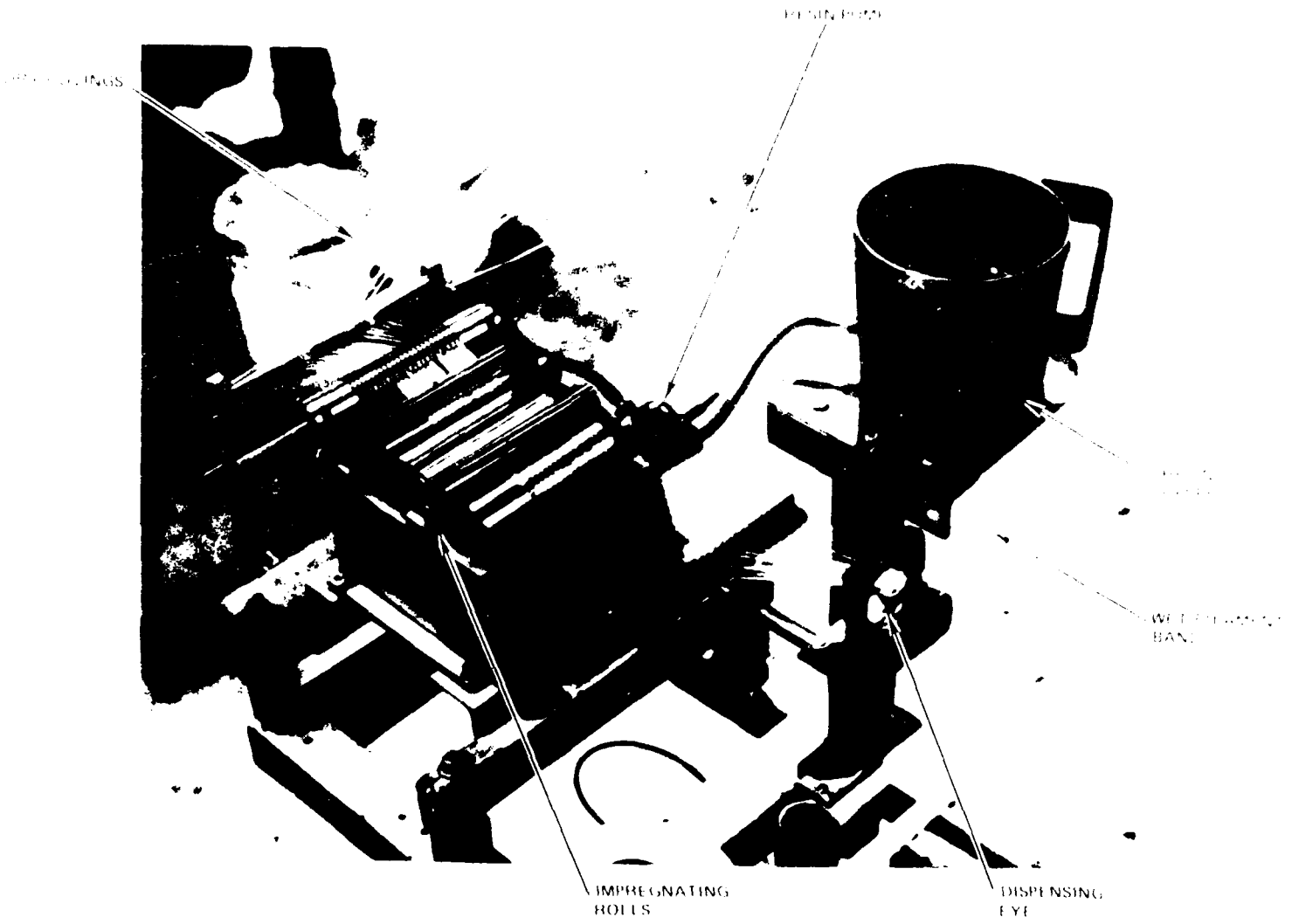


Figure 53. Resin Impregnator

AD-A121 172 MANUFACTURING METHODS AND TECHNOLOGY (MANTECH) PROGRAM MANUFACTURING TECH..(U) HUGHES HELICOPTERS INC CULVER CITY CA J V ALEXANDER ET AL. OCT 81 HHI-81-367 2/3

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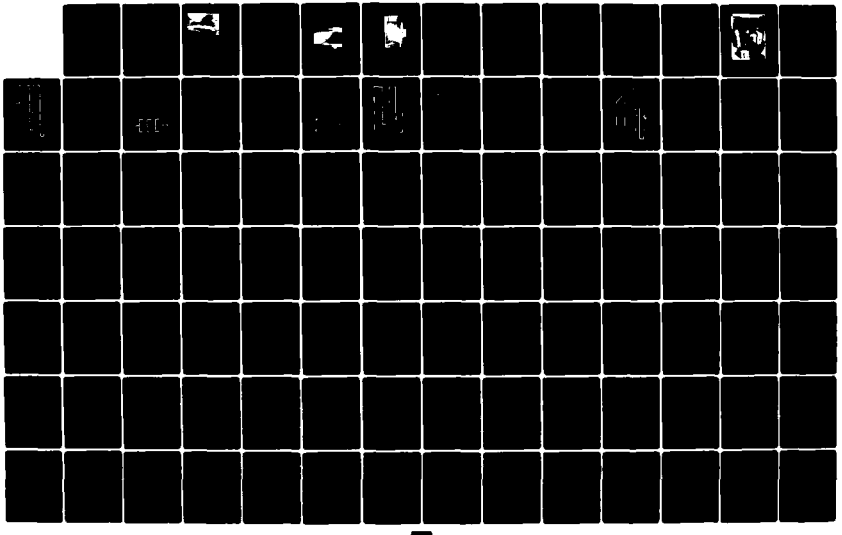
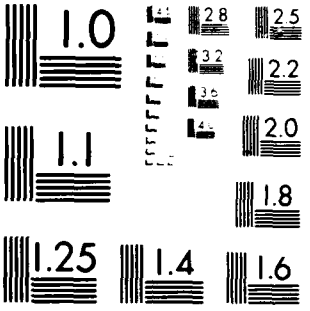




TAILBOOM

The tailboom is a sandwich wall, hollow cone with the honeycomb filler deleted at both ends of the structure to make the walls a solid laminate in these regions. The configuration of the mandrel on which the tailboom would be wound is indicated in Figure 59. From Sta 370 to 450 the diameter tapers and the cross section varies smoothly from an essentially rectangular shape at the front to a circle at Sta 450. It is a circular cone between Stations 450 and 529, and then is cylindrical to the end of the tailboom. The shape of the end dome on the large end of the mandrel is critical for guiding the filament turnaround and also to guard against the ±8 degree filaments bridging over the flat region on the top of the forward end of the tailboom. The mandrel's geometry was worked out on the full-scale toolproofing mandrel shown in Figure 60. (This was lightweight fiberglass mandrel adequate for the development work but not suitable for CTS fabrication.) Figure 61 shows the removable segment that aligned the four intercostals that lie in the aft end of the tailboom, and the end domes. The ring frames that are centered on Sta 532.33 are clamped between the right end of the mandrel shown in Figure 60 and the left end of the slotted mandrel shown in Figure 61. The three ring frames (Sta 387.00, 531.23, and 533.43) and the four intercostals are hand layups. They are made in metal female molds to provide accurate contours for matching the interior of the tailboom, using silicon blocks to supply the molding pressure. This technology was considered to be well enough understood that it was not necessary to undertake a special manufacturing development program.

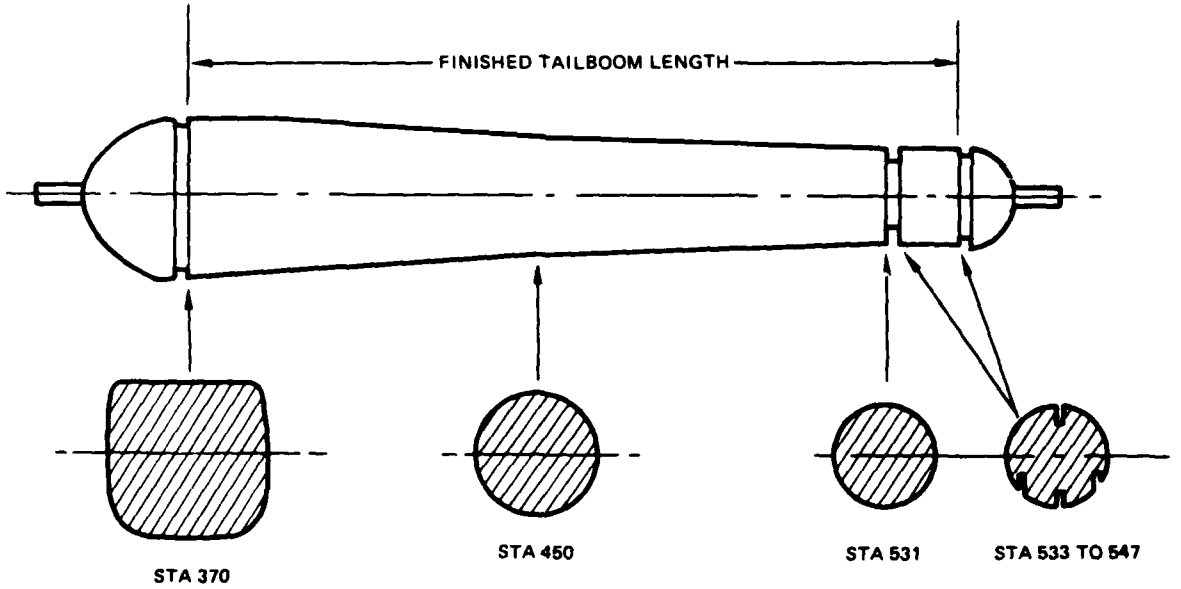


Figure 59. Tailboom Mandrel Geometry

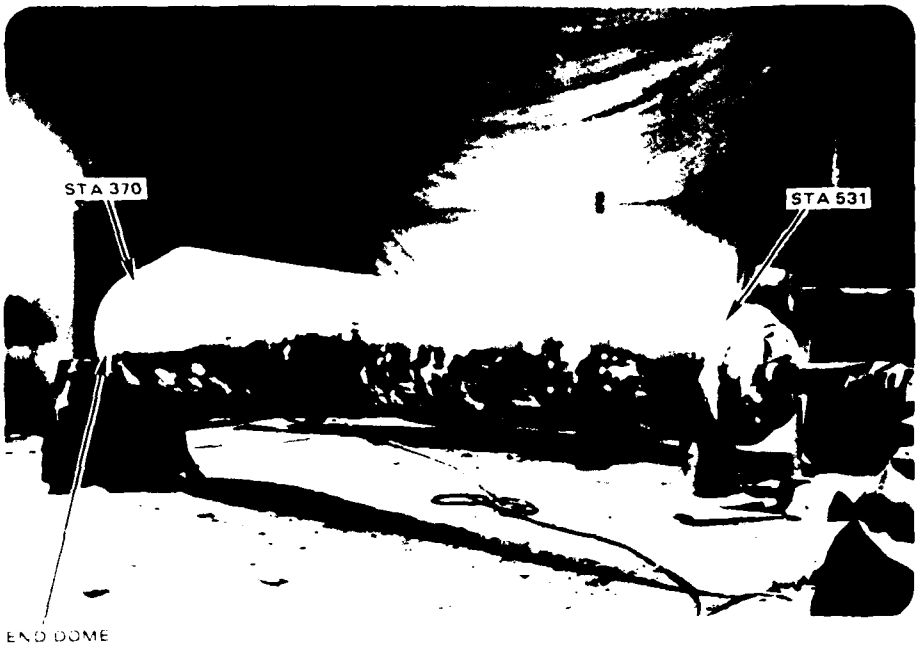


Figure 60. Tailboom Mandrel - Main Section (Toolproofing)

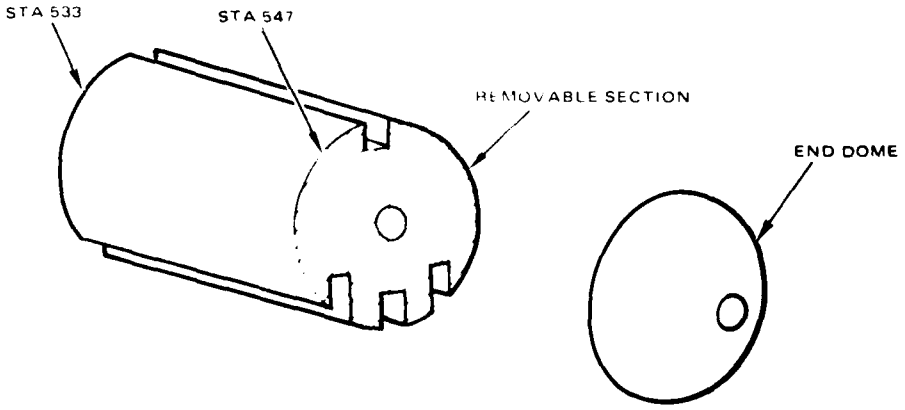


Figure 61. Tailboom Mandrel - Removable Sections (Toolproofing)

VERTICAL TAIL SPAR

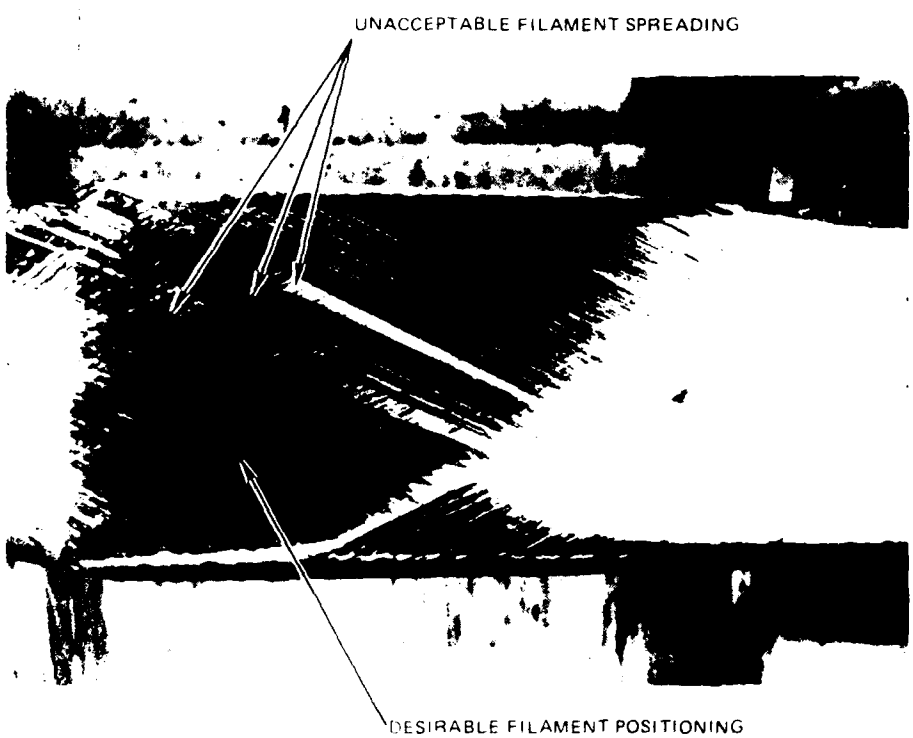
The structure of this spar is similar to that of the tailboom - solid laminate walls at the ends and sandwich wall in the center. The major difference is that its periphery is smaller, and its cross section is generally trapezoidal in shape. This spar, too, has frames for the tail rotor gearbox support wound into the spar.

A full-scale toolproofing mandrel was made of fiberglass and was used for determining the winding procedure (see Figure 60). The development work included determining the manner of holding the composite frames in place in the mandrel, developing the configuration of the end domes, and determining how to remove the mandrel from the cured spar box. Figure 62 shows the results of winding on an early, unsatisfactory end dome configuration. Note how the filaments are spread in an unacceptable manner.

The frames that go into the vertical tail spar were proposed to be made in female metal molds with silicone blocks for pressurization. This technology was considered to be well enough understood that toolproofing studies were not needed.

STABILATOR

An early stabilator configuration had two spars, four ribs, and a bead-stiffened skin. A tooling development study was made of the skin which is shown in Figure 63. HHI had previous experience with a beaded skin for a composite stabilator for the OH-58A in which a flat, wet skin was worked onto an airfoil-shaped mandrel that had ridges to form the beads. While this was successful for the small size of the OH-58A, it was determined to be inappropriate for the AH-64A stabilator because of the size, and because of the filament distortion that was caused by forcing the flat skin to form around the deep beads. A second step, which is shown in Figure 63, consisted of forming beads separately and cocuring them to the skin that was draped over a single curvature mold. Based on the toolproof skin, this too appeared impractical from an overall skin stability standpoint and the design concept was changed to the spar/rib/skin structure described in the Design Refinement section. The technology for forming the spars and ribs in female metal molds with silicone blocks for pressure was considered well enough known not to need special development.



. Vertical Tail Spar Winding Pattern Study

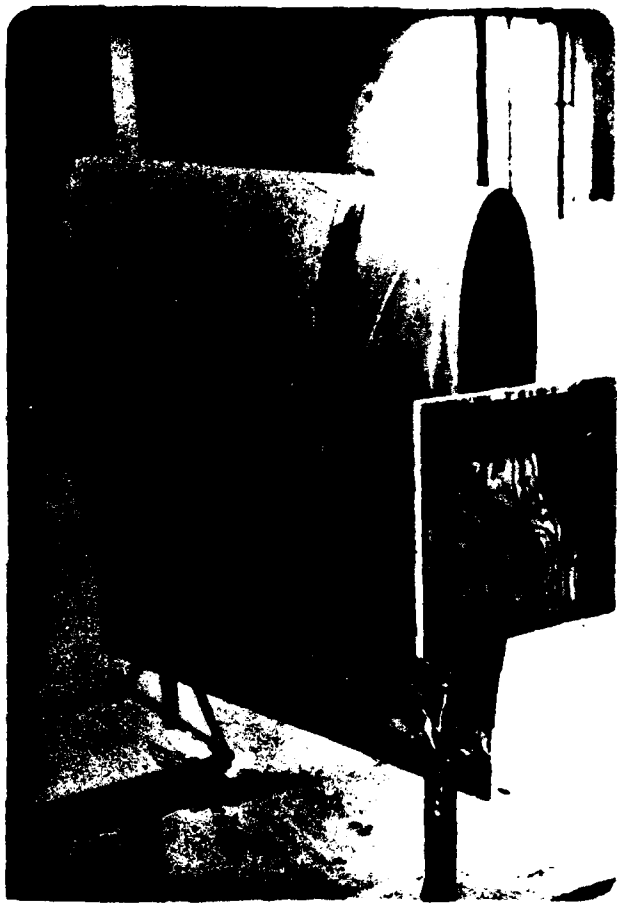


Figure 63. Bead-Stiffening Study for Stabilator Skin

RESIN CURE CYCLE

HHI and FSI recommended that Applied Plastics Corporation resins and hardeners be used as the epoxy matrix for the structure:

- 2434 – epoxy resin
- 2347 – 300°F cure hardener
- 2340 – room temperature cure hardener

The 300°F cure hardener, 2347, would be used for all filament winding of major structural elements. The 2340 hardener would be used for small, noncritical parts. The 2347 hardener is formulated to have a 24-hour pot life at room temperature. This time can be extended to 72 hours if stored at temperatures below 40°F, and to two weeks if stored at 0°F.

The cure cycle for the APCO resins has been determined through evaluation tests to be as shown in Figure 64. The Phase I cure is analogous to "B-staging" in conventional prepregs. This cycle rigidizes a component by partially cross-linking the resin so that the component is rigid enough to be handled, mounted in winding mandrels so wet filaments may be wound over it, or positioned in assembly fixture for final assembly by cocuring.

The "final assembly by cocuring" cycle that goes to 300°F raises the glass transition temperature above the range of normal operating temperatures for strength at elevated temperatures, and produces a structure that is free of micro-cracking.

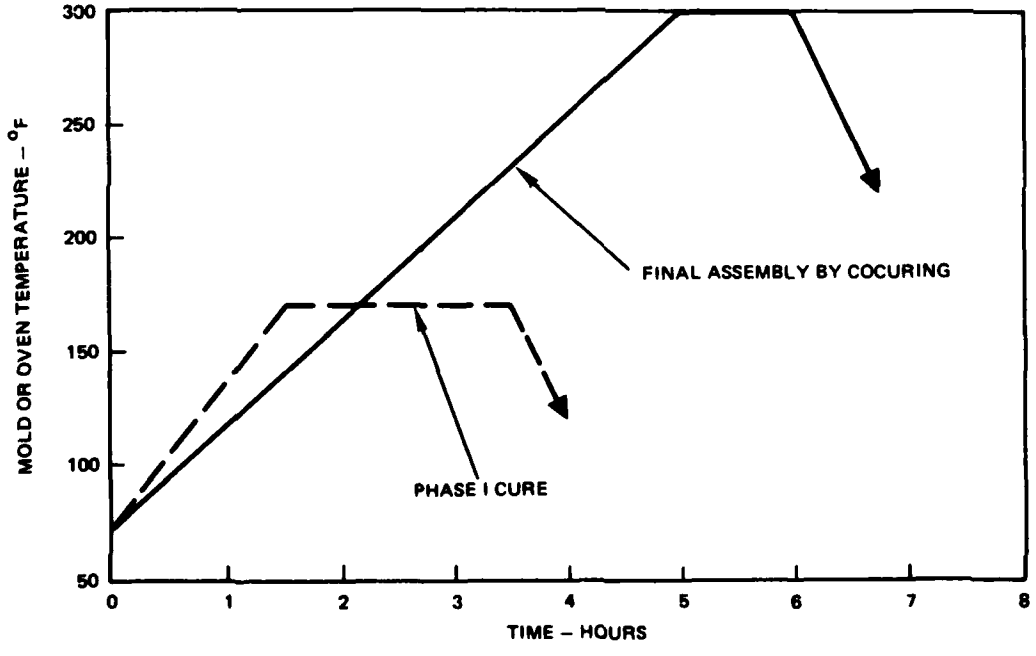


Figure 64. Resin Cure Cycle

FABRICATION REFINEMENT

At the time that the program was terminated, no production tooling had been built. The concept and approximately 60 percent of the drawings for it had been completed. The three components of the CTS were to be built on aluminum tooling to fulfill the requirement that the tools be satisfactory for fabricating a minimum of 50 shipsets of the CTS. The jig for assembling and aligning the three CTS components as an assembly was to use the tailboom winding mandrel as the base, with jigs and fixtures added to support the various components for bonding or bolting them together.

TAILBOOM TOOLING

The proposed tailboom mandrel is shown schematically in Figure 65. It consists of five major parts, plus two clamps to lock it together. The main section is generally conical in shape and has a shaft extending out of each end. It is shaped to the inside contour of the tailboom from Sta 370 to 429 and is an aluminum monocoque structure stiffened with bulkheads and longerons. The shafts are supported by the two bulkheads nearest to each end. The shafts and the bulkheads are perforated so that hot air may be circulated inside the mandrel while the resin cures in the oven.

The large end turn-around dome slips over the shaft of the main mandrel and is pinned to it so they will turn together. A wedge-type clamp anchors the dome to the shaft. The dome is contoured to guide the filaments smoothly as they leave and reapproach the mandrel, and to prevent bridging of the filaments across the flat spot on the mandrel. The dome has a channel all around it where it touches the main mandrel to serve as a cutoff guide for trimming the cured tailboom. This channel is to be filled with plastic foam during the winding process to present a smooth winding surface.

The cylindrical segment adjacent to the small end of the main mandrel slips over the shaft, is pinned to the main mandrel, is held against the main mandrel by the small end dome, and is locked to the shaft by a wedge lock at the outboard end of the end dome. This cylindrical mandrel is spaced away from the end of the main mandrel far enough to enclose the two graphite/epoxy frames that are bonded together as an assembly with aluminum tooling blocks between, and are centered at Sta 532.33. The tooling for this assembly is indicated in Figure 66. The cylindrical section of the mandrel has four lengthwise channels formed into its surface to accept the four graphite/epoxy intercostals that lie in the aft end of the tail boom. The end dome has a channel around its periphery where it touches the cylindrical mandrel to serve as a cutoff guide. This channel is filled with foam to provide a fair surface for winding.

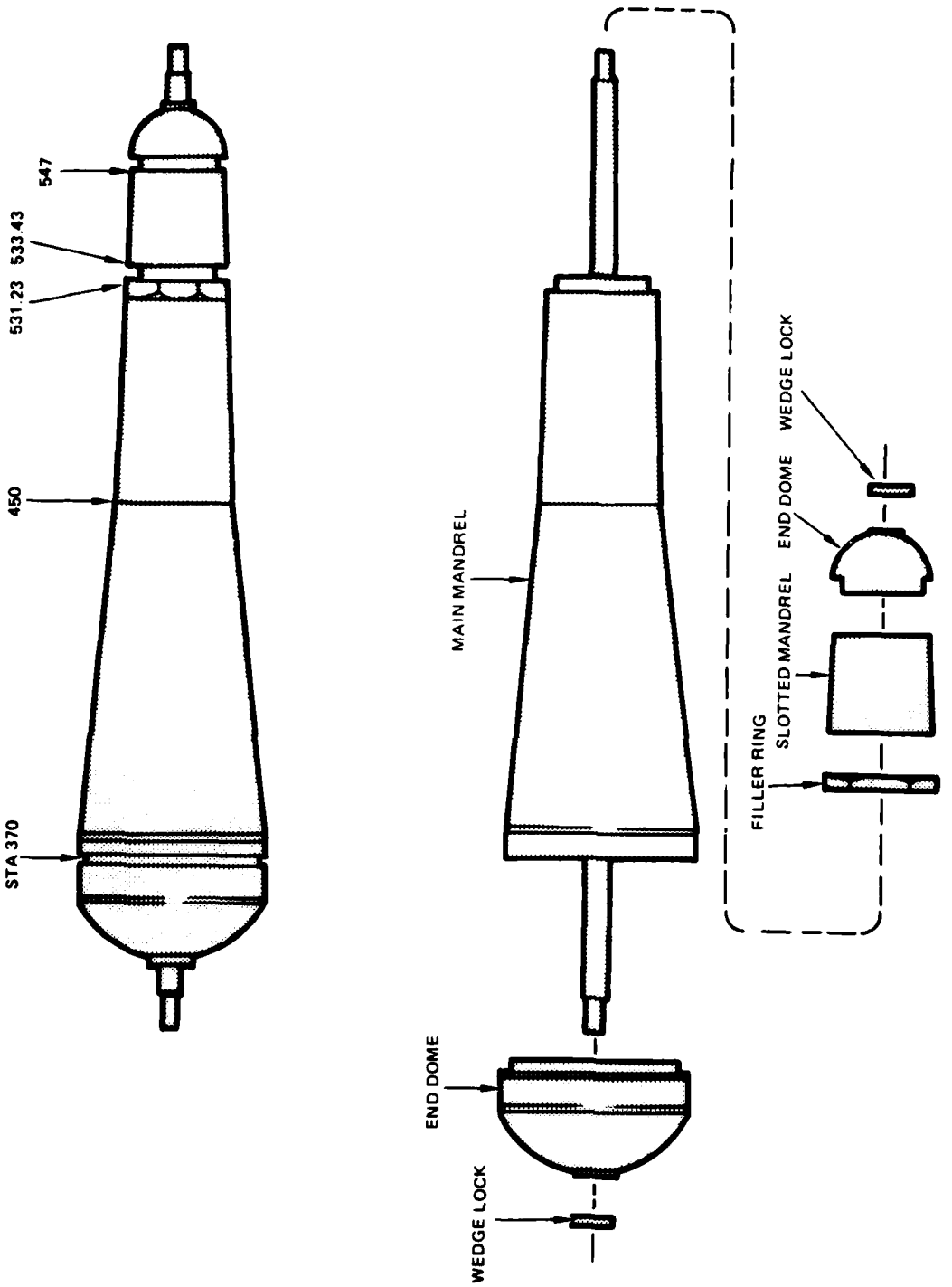


Figure 65. Tailboom Mandrel

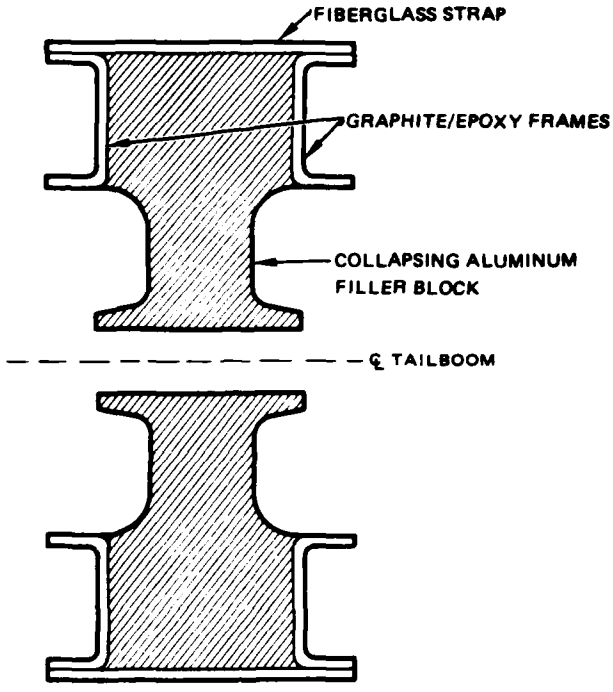


Figure 66. Frame Assembly Fixture
Schematic

The filler ring indicated in Figure 65 is a removable, tapered aluminum ring that fits closely onto the main mandrel and is shaped to the tapered aft end of the honeycomb filler. This ring must be able to slip off of the aluminum mandrel so that the cured tailboom will not be mechanically locked to the mandrel.

The graphite/epoxy frames at Sta 387.66, 531.23, and 533.43 and the four graphite/epoxy intercostals are formed in matched metal dies using trapped silicone rubber to furnish the molding pressure during the cure cycle. Figure 67 is a schematic cross section of such a mold. The design shows the Sta 533.43 frame and the intercostals attached to each other by HyLoc fasteners. This subassembly is located in the mandrel as the mandrel is being assembled.

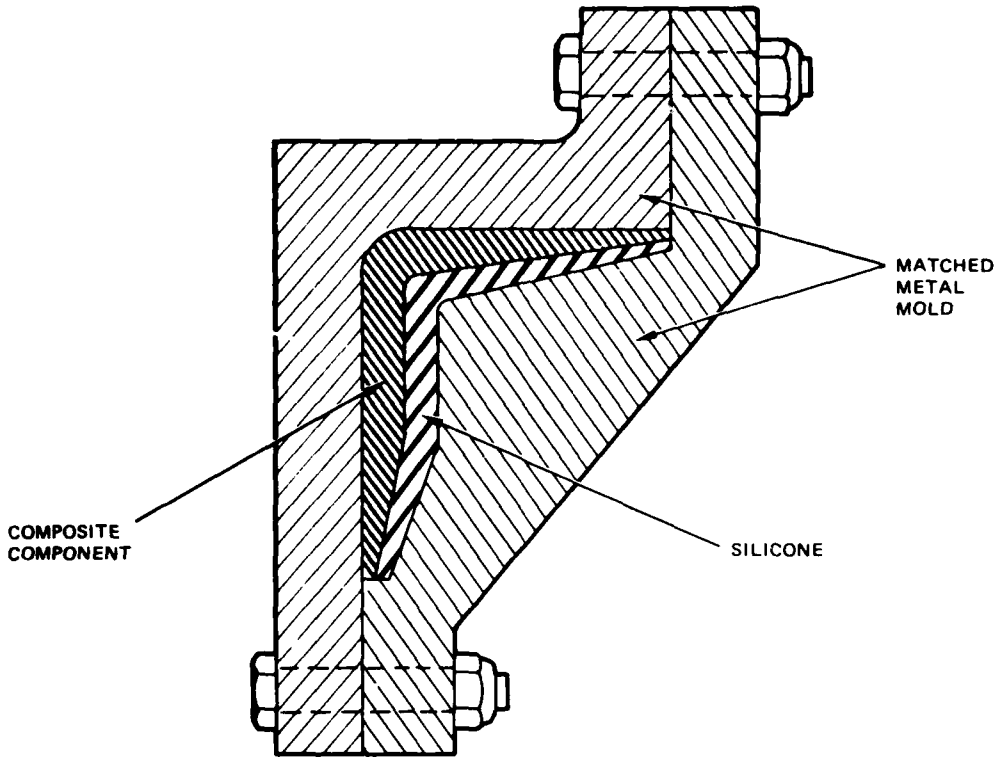


Figure 67. Frame Mold Section Schematic

Flat-pattern metal templates are provided for cutting out the honeycomb filler for the sandwich portion of the tailboom, and for the individual sheets of prepreg material that are laid up to form the ring frames, intercostals, and tailboom skin reinforcements. Metal jigs that key to the mandrel serve to locate the reinforcements.

A conventional router is set up to bevel the ends of the honeycomb panels.

A wet filament winding machine similar to the one shown in Figure 68 is used to wet filament wind the tailboom. This machine takes dry composite rovings, impregnates them with a metered amount of resin, and winds them onto the mandrel at the spiral angles required by the design.

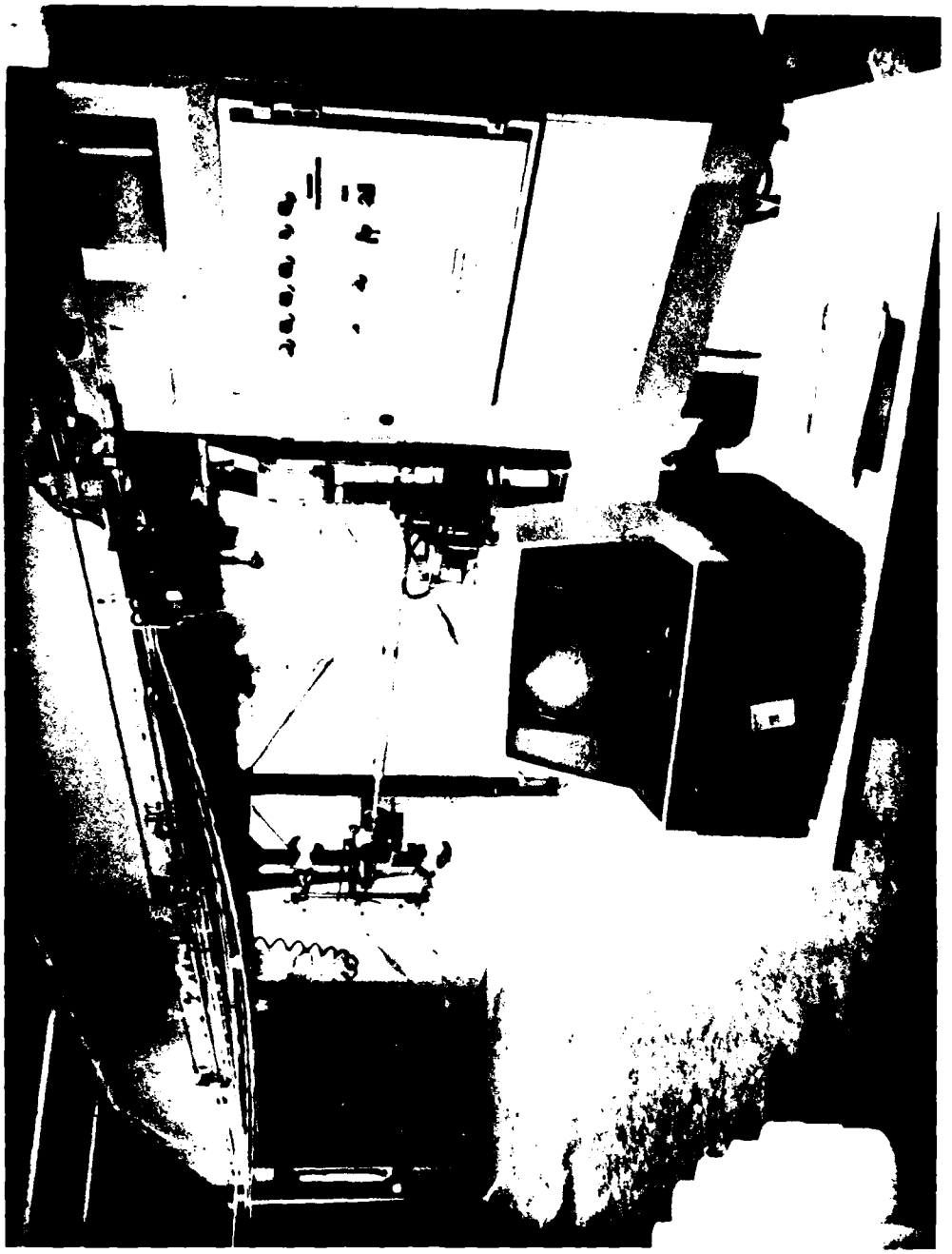


Figure 68. Helical Winding Machine

Aluminum male molds (Figure 69) are used to lay up prepreg material to form the angles that attach the tail rotor driveshaft cover.

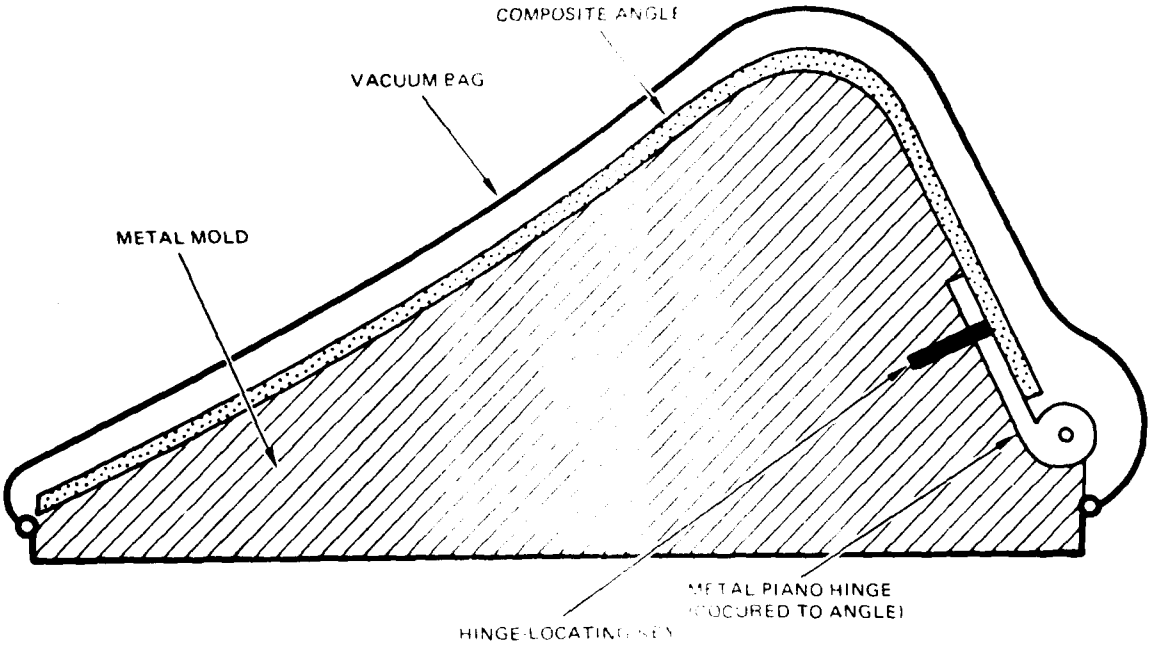


Figure 69. Angle Mold Schematic

The proposed sequence for fabricating the composite components of the tailboom is laid out in Figure 70. It begins with the fabrication of premolded frames, goes through filament winding, curing, and subassembly of miscellaneous parts. From this operation it goes on to final assembly that is described below.

VERTICAL TAIL TOOLING

The mandrel for the hollow box structure of the vertical tail spar is the same in concept as that for the tailboom, except that it has a trapezoidal cross section. See Figure 71. The main part of the mold is an aluminum monocoque box with a shaft extending from each end. This mandrel is formed to the inside contour of the spar box. An end dome that is shaped to

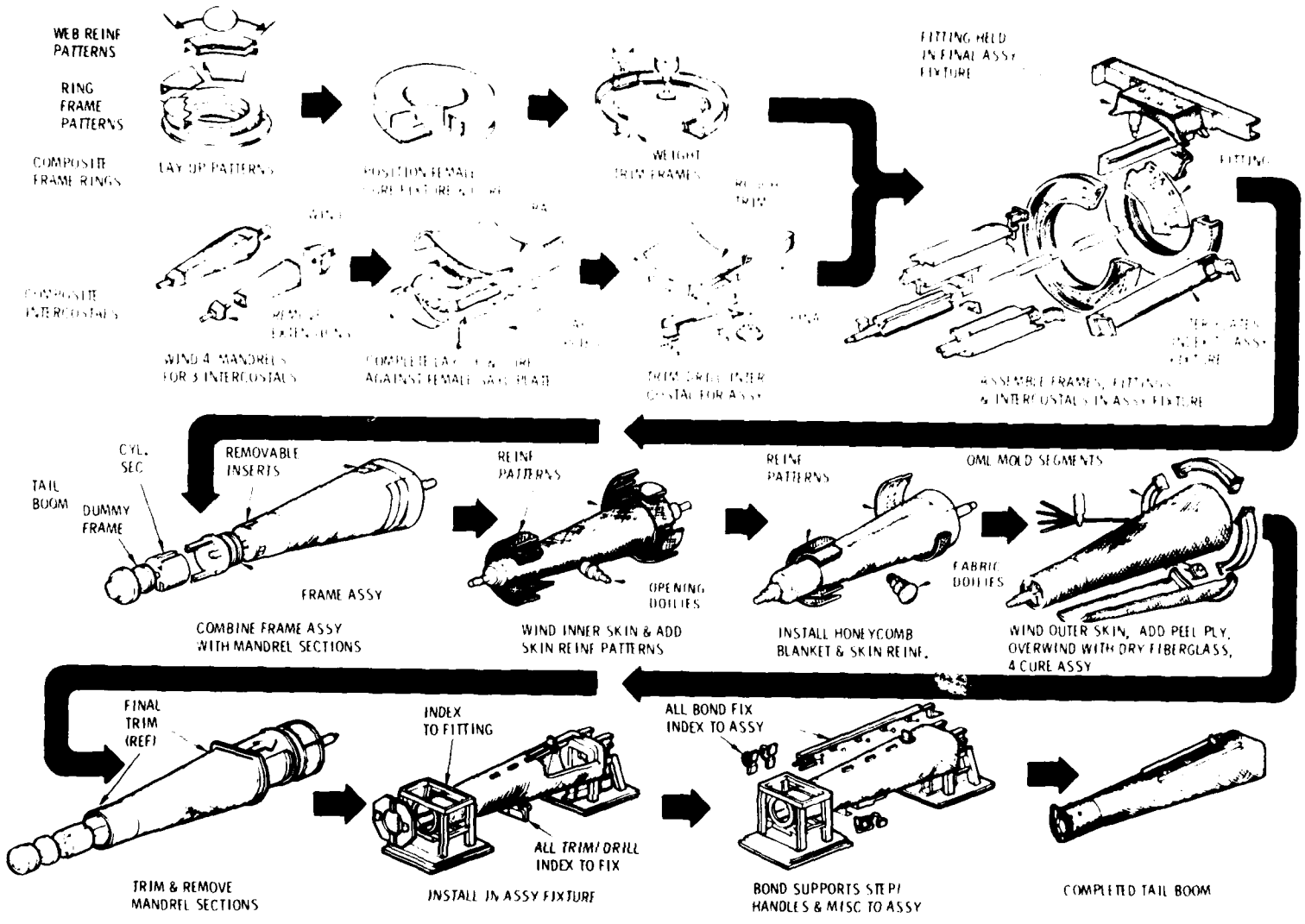


Figure 70. Tailboom Fabrication Sequence

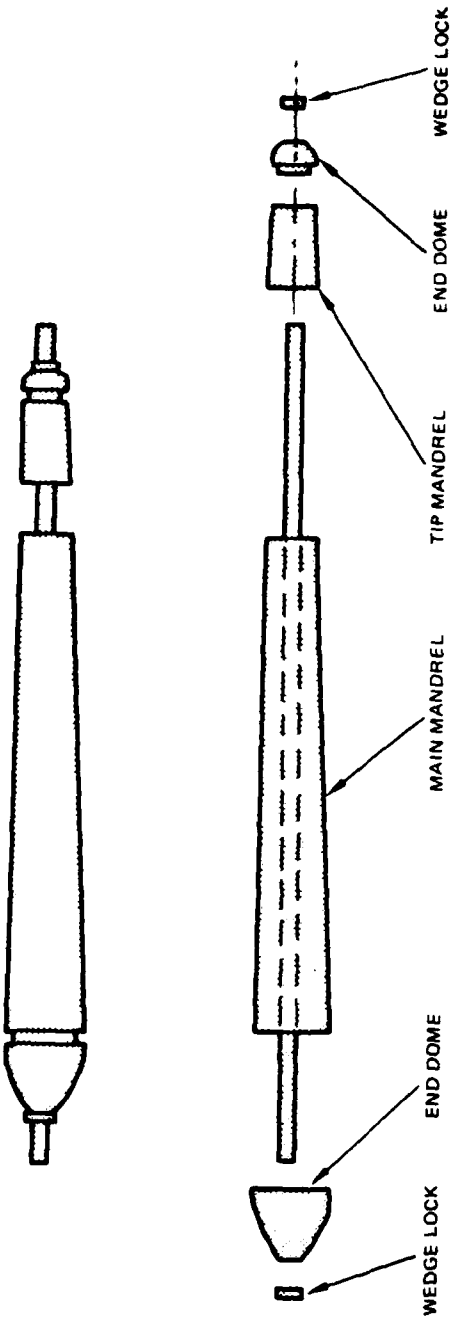


Figure 71. Vertical Tail Spar Mandrel

perform the filament turnaround function and guide the filaments off of and onto the four-sided mandrel slips over the extension shaft at the big end of the mandrel. It is pinned to it to force them to rotate together and is secured by a wedge-type clamp. The tip end portion of the mandrel slides over the main mandrel's extension shaft and is pinned to it. A suitable end dome is mounted beyond this portion of the mandrel and is attached by another clamp. The tip-end portion of the mandrel is spaced away from the upper end of the main mandrel to allow insertion of the upper frame subassembly so that it may be wet filament wound into the vertical tail spar. Provisions are made for keying this subassembly to the mandrel.

The five composite frames (three upper frames to support the tail rotor gearbox and two lower frames to support the intermediate gearbox) are formed in matched metal molds with trapped silicone rubber furnishing the molding pressure.

The three frames that support the tail rotor gearbox are formed into an assembly that goes into the spar box mandrel. A three part wash-out mold locates the three frames while a prepreg shell is wrapped around them and cured. See Figure 72.

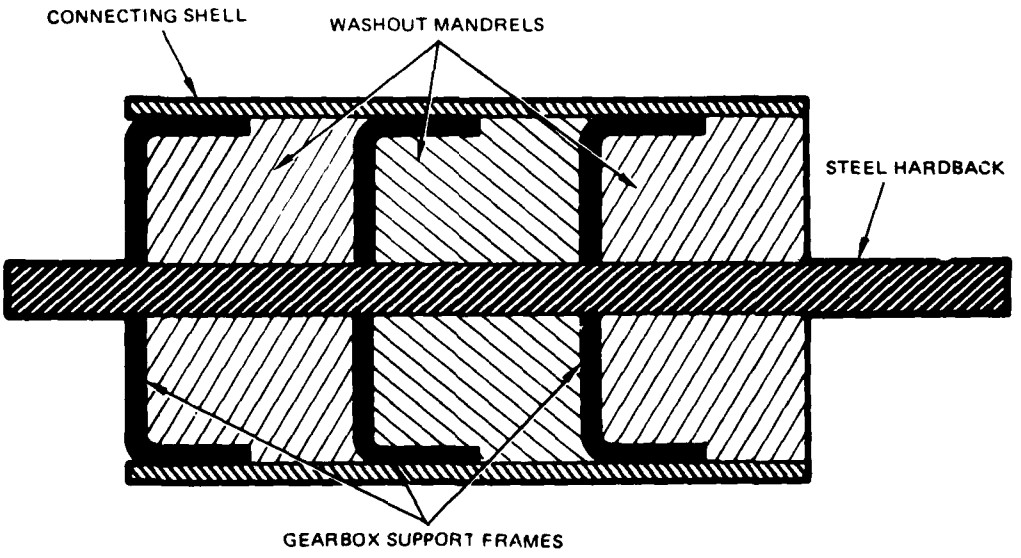


Figure 72. Frame Subassembly Tool Schematic

Flat pattern metal templates are used to cut out the honeycomb filler for the spar box, and for the laminates that make up the frames, doublers, and fairings.

A routing fixture is clamped to the spar box and the top and bottom ends of the box are cut to shape. A support fixture holds the spar box and provides jig points for positioning the two lower bulkheads, and the four aluminum lower attachment fittings. These are held in place while HyLoc fasteners secure the components. A conventional milling machine faces off the attachment lugs and precisely bores holes in them for the vertical tail attachment bolts.

A female metal mold is used to form the vertical tail's tip cap, and contains appropriate keys for locating the navigation lights, radar warning antennas, and lightning protection strip. It is shown schematically in Figure 73.

The trailing edge fairing is a purchased part that contains communications antennas and is furnished by the antenna subcontractor. The leading edge fairing is taken unchanged from the prototype vertical tail.

A closed cavity metal mold with silicone rubber pressurizing blocks is used for forming each of the T-bars that are placed at the corners of the spar box for attaching the leading and trailing edge fairings. Figure 74 is a schematic cross section of such a mold.

The T-bars are bonded to the four corners of the vertical tail spar in a bonding jig similar to that shown in Figure 75. Trapped silicone rubber is used to supply the bonding pressure.

Figure 76 shows the proposed sequence for fabricating the vertical tail spar by the wet filament winding process. This spar assembly goes into the CTS final assembly process as described below.

STABILATOR TOOLING

The stabilator skin is wet filament wound on a cylindrical aluminum mandrel (Figure 77), cut off, and cured on an aluminum male mold shaped to the airfoil contour as Figure 78 indicates.

The spars and ribs are laid up in matched metal molds with silicone rubber pressurization in a manner similar to that described for the tailboom frames in Figure 67.

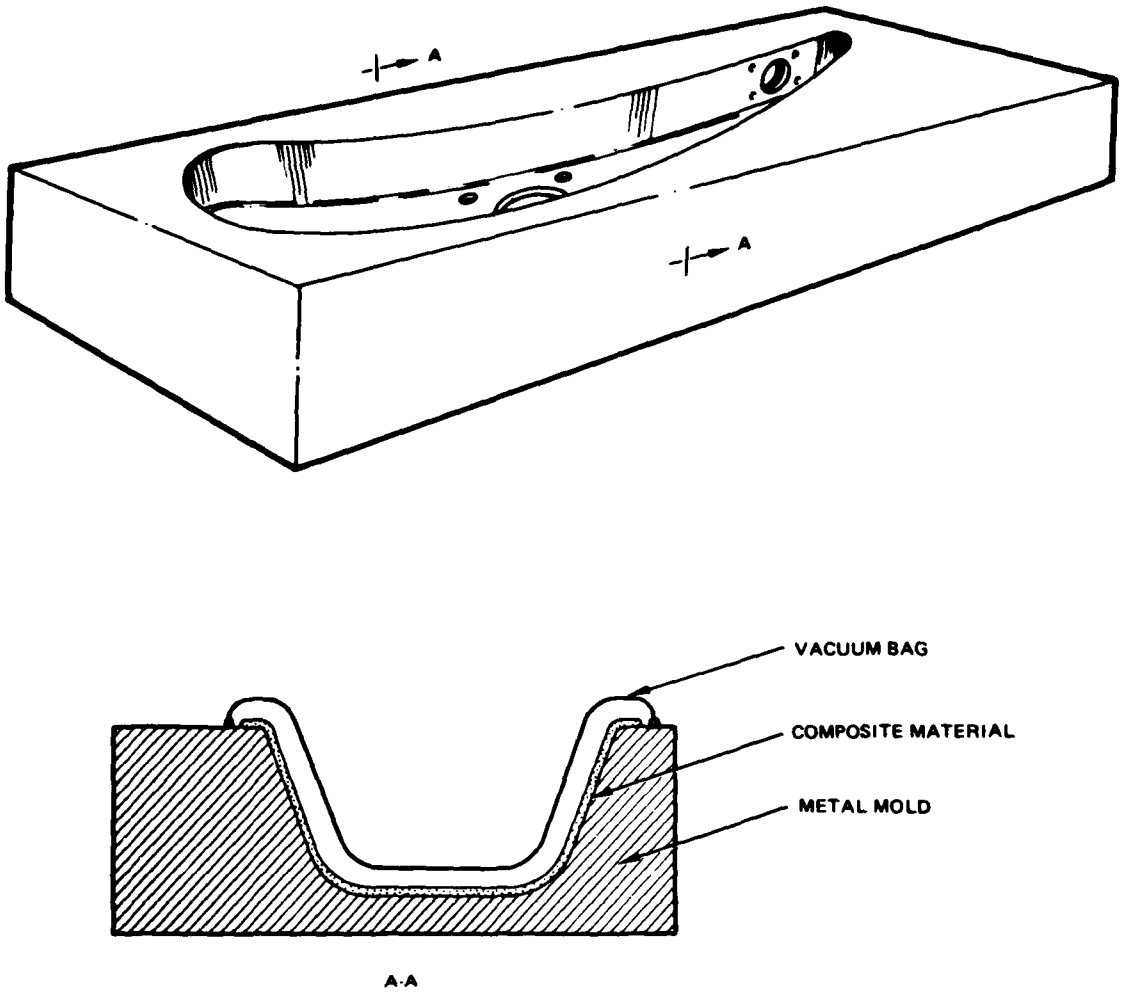


Figure 73. Tip Cap Mold Schematic

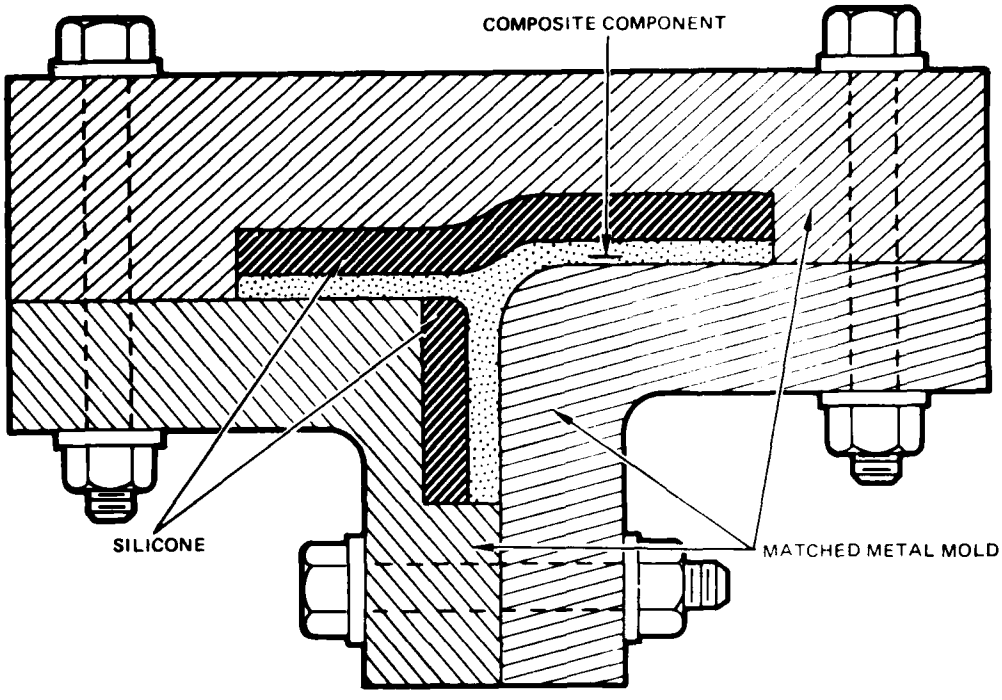


Figure 74. T-Bar Mold Section Schematic

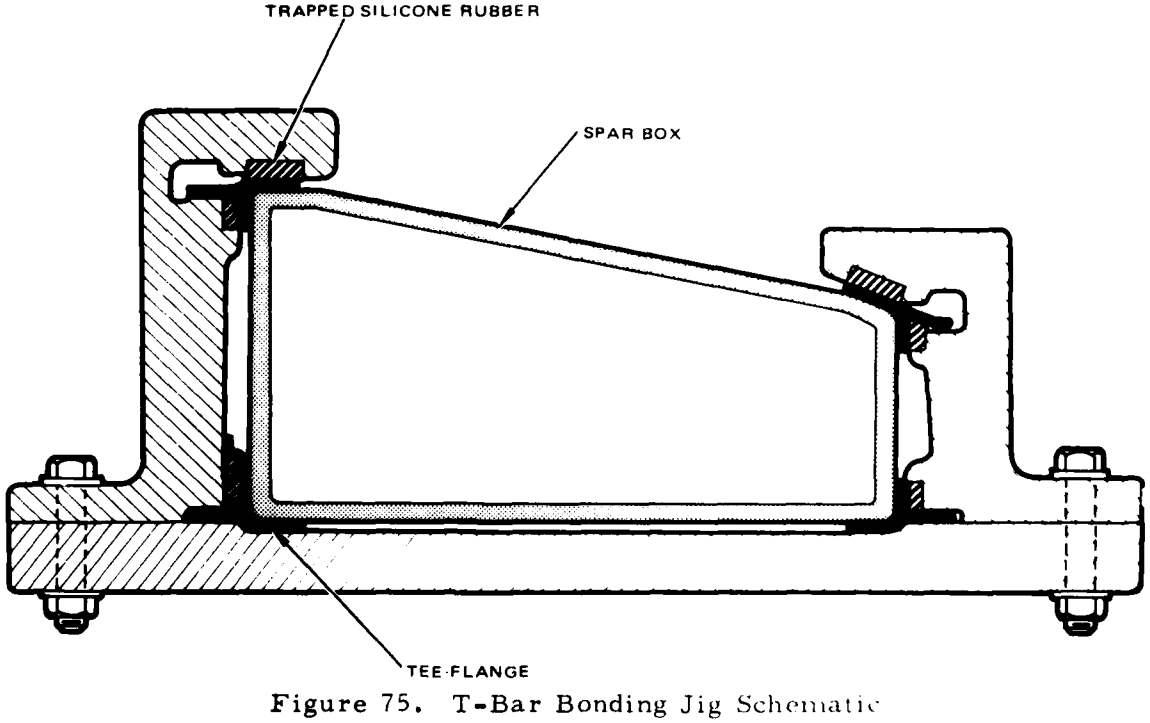


Figure 75. T-Bar Bonding Jig Schematic

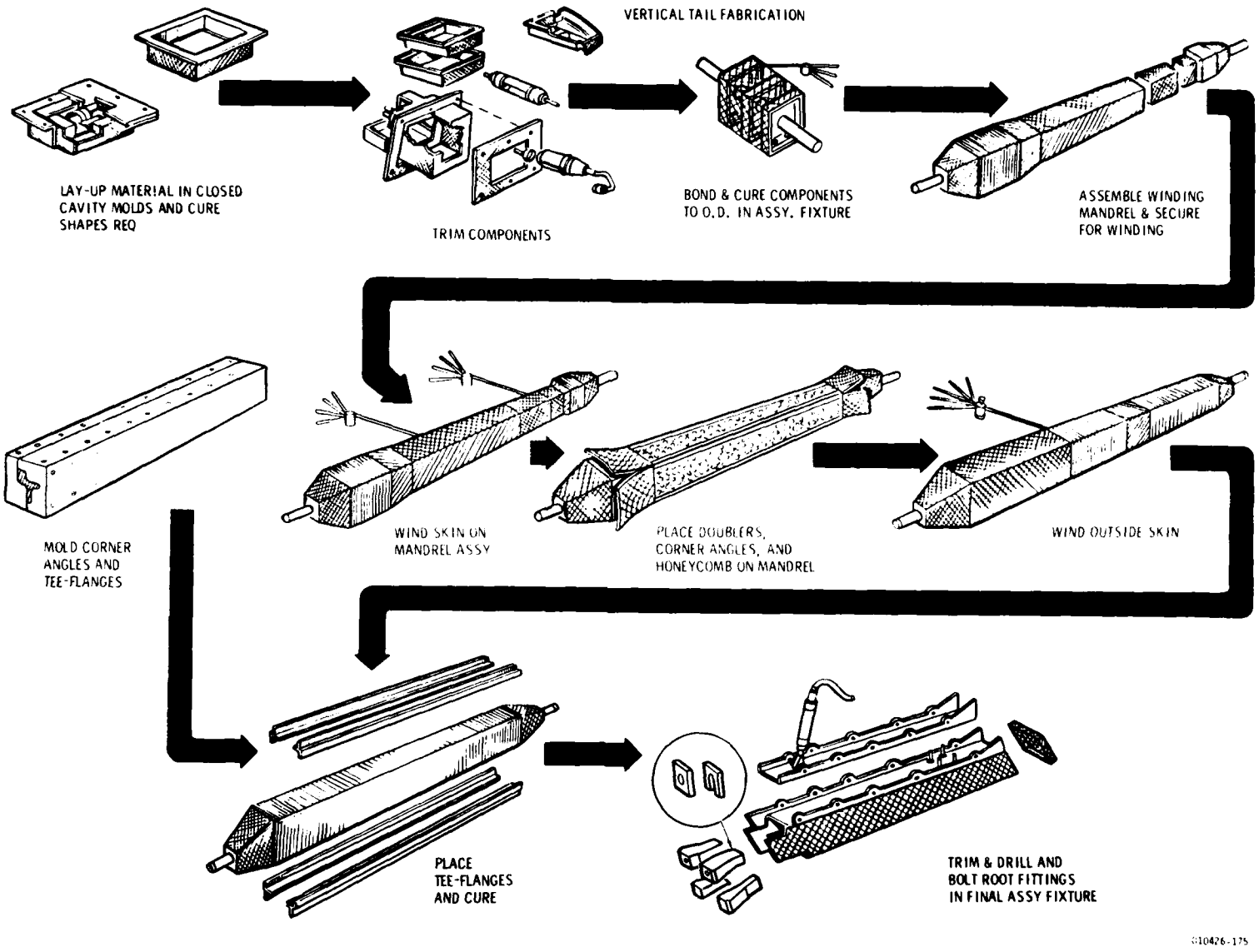


Figure 76. Vertical Tail Fabrication Sequence

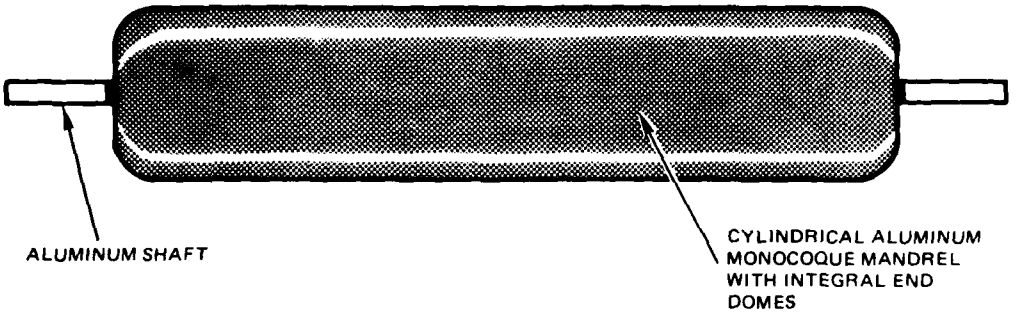


Figure 77. Stabilator Skin Winding Mandrel Schematic

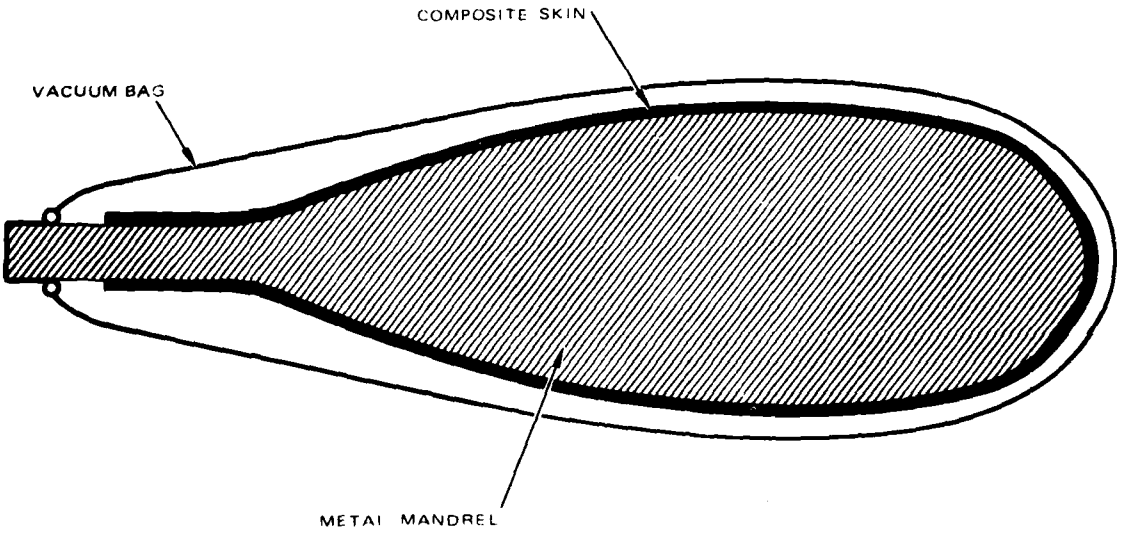


Figure 78. Stabilator Skin Mold Schematic

The tip caps are laid up on metal male mold as shown in Figure 79.

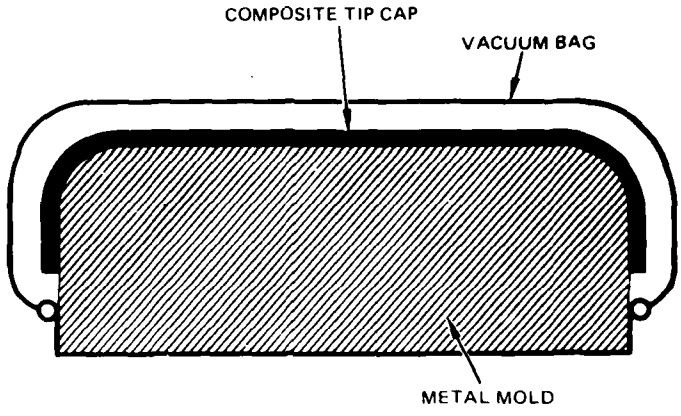


Figure 79. Tip Cap Mold Schematic

The stabilator's interior structure is assembled in a fixture that locates the spars, ribs, and tip caps. It includes clamping devices to apply bonding pressure where the ribs and tip caps bear against the spar. The stabilator final assembly jig (Figure 80) is basically a grid structure that locates the skin segments over the internal structure of the stabilator and has an array of elastomeric pads that provide the bonding pressure. They press the skin against the spars and ribs, and the upper and lower skins together at the trailing edge. A single blind rivet at the nose of each rib aligns the skin and ribs positively. This jig also has provisions for aligning the stabilator hinge fittings with respect to the stabilator, so that they may be attached with mechanical fasteners.

The process shown in Figure 81 is the proposed fabrication sequence for the stabilator. Final assembly into the CTS is described below.

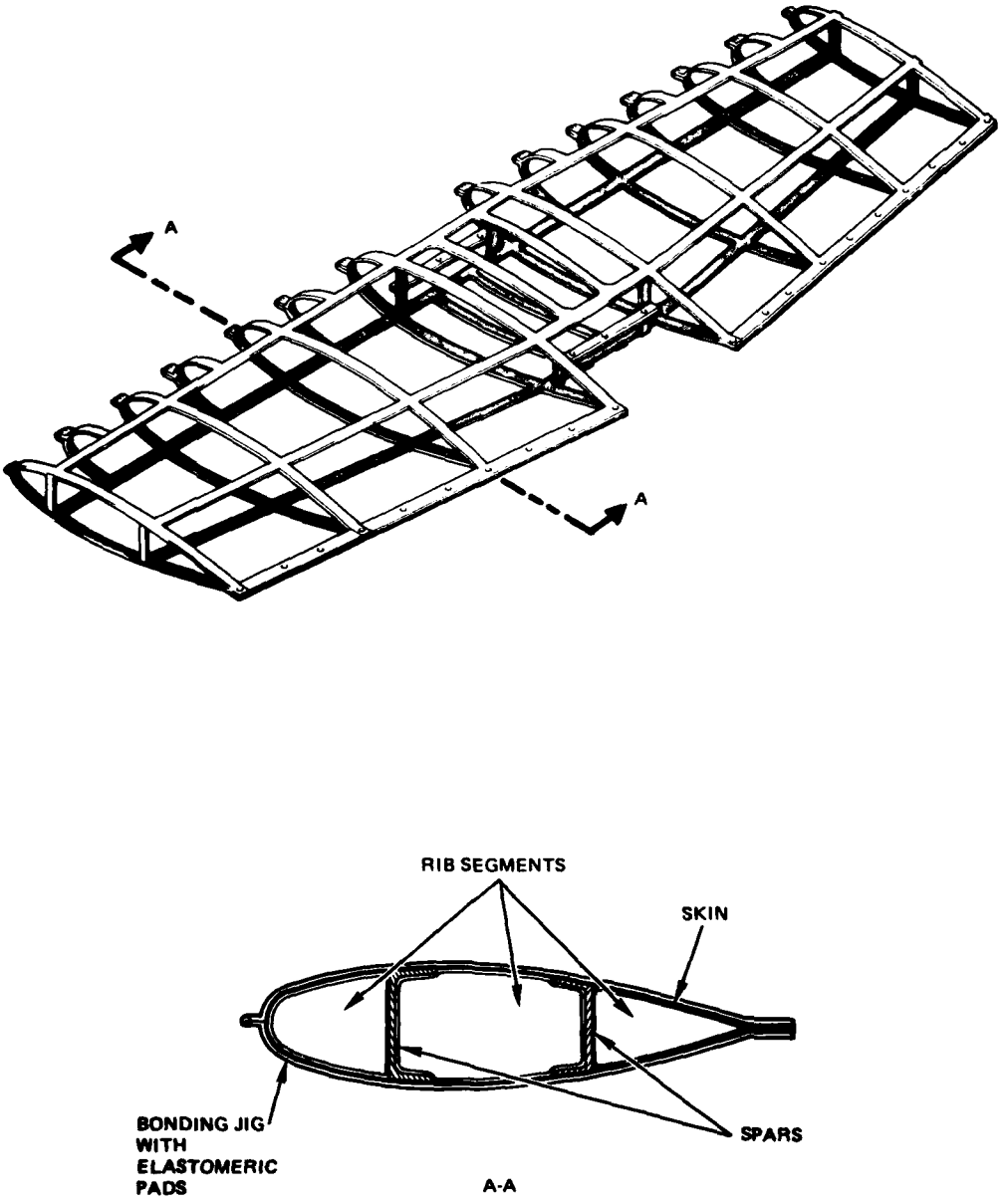


Figure 80. Stabilator Skin Bonding Jig Schematic

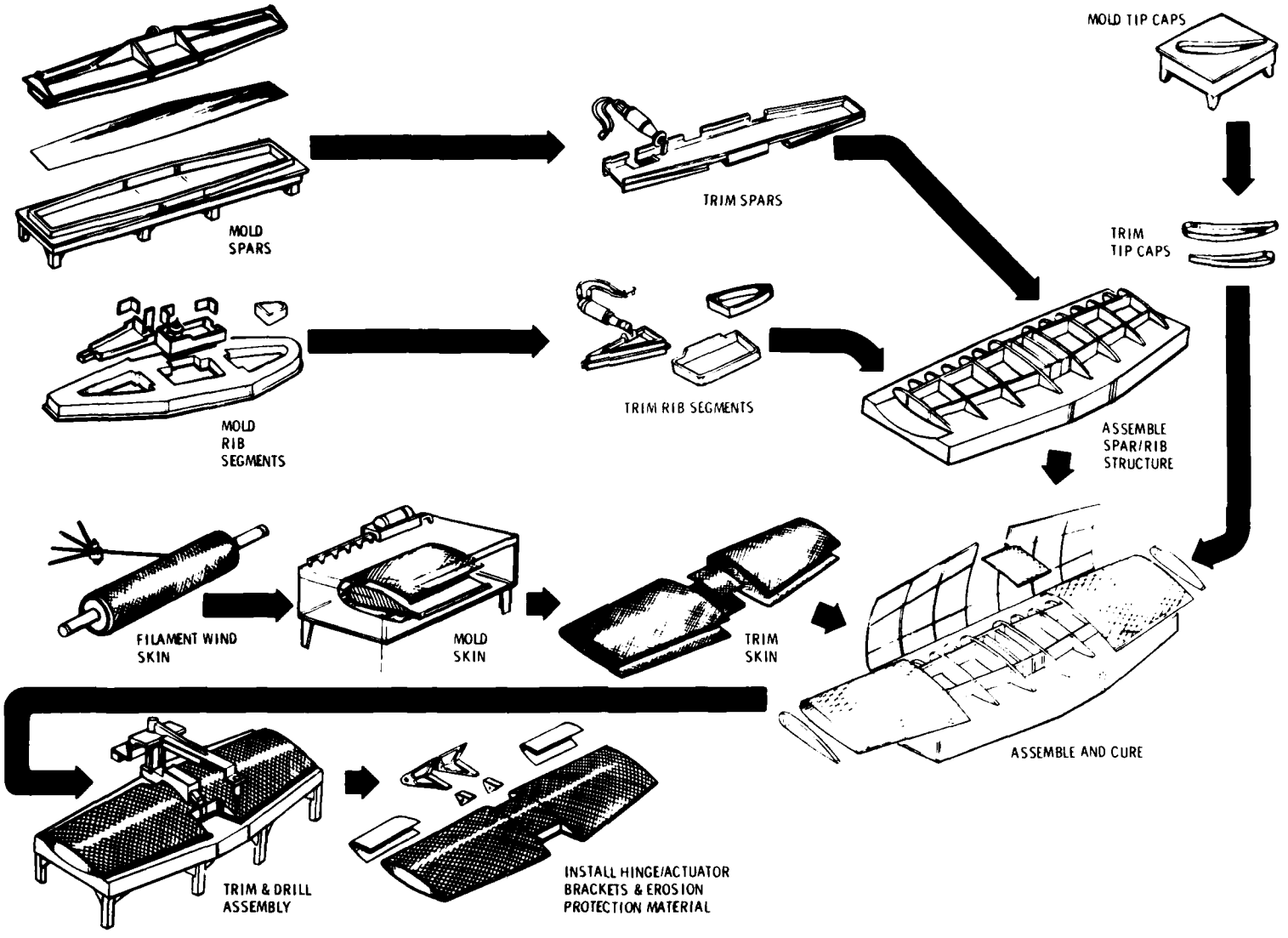


Figure 81. Stabilator Fabrication Sequence

FINAL ASSEMBLY

Final assembly of the CTS is accomplished in the fixture that is shown schematically in Figure 82. Its first purpose is to locate the aluminum rear bulkhead in the tailboom and hold it while it is bolted in place. Next it serves as a jig for drilling the four bolt holes that attach the vertical tail, for milling off the top of the vertical tail attach bosses, and for locating the jacking/tiedown fitting. It has drill jig plates for locating the tail rotor driveshaft bearing and anti-flail supports and for mounting the steps and flare dispenser. It holds the vertical tail in alignment while the gearboxes and controls are installed. It positions the stabilator while its actuators are adjusted.

When the CTS assembly is taken out of this fixture, it is ready to be mounted on the AH-64A airframe. An alignment jig is needed for attaching the CTS to the metal fuselage at Sta 370; a jig that is not a part of the MM&T program tooling but is to be provided by the AH-64A project organization under AVRADCOM Contract DAAJ01-77-C-0064, Modification P00108.

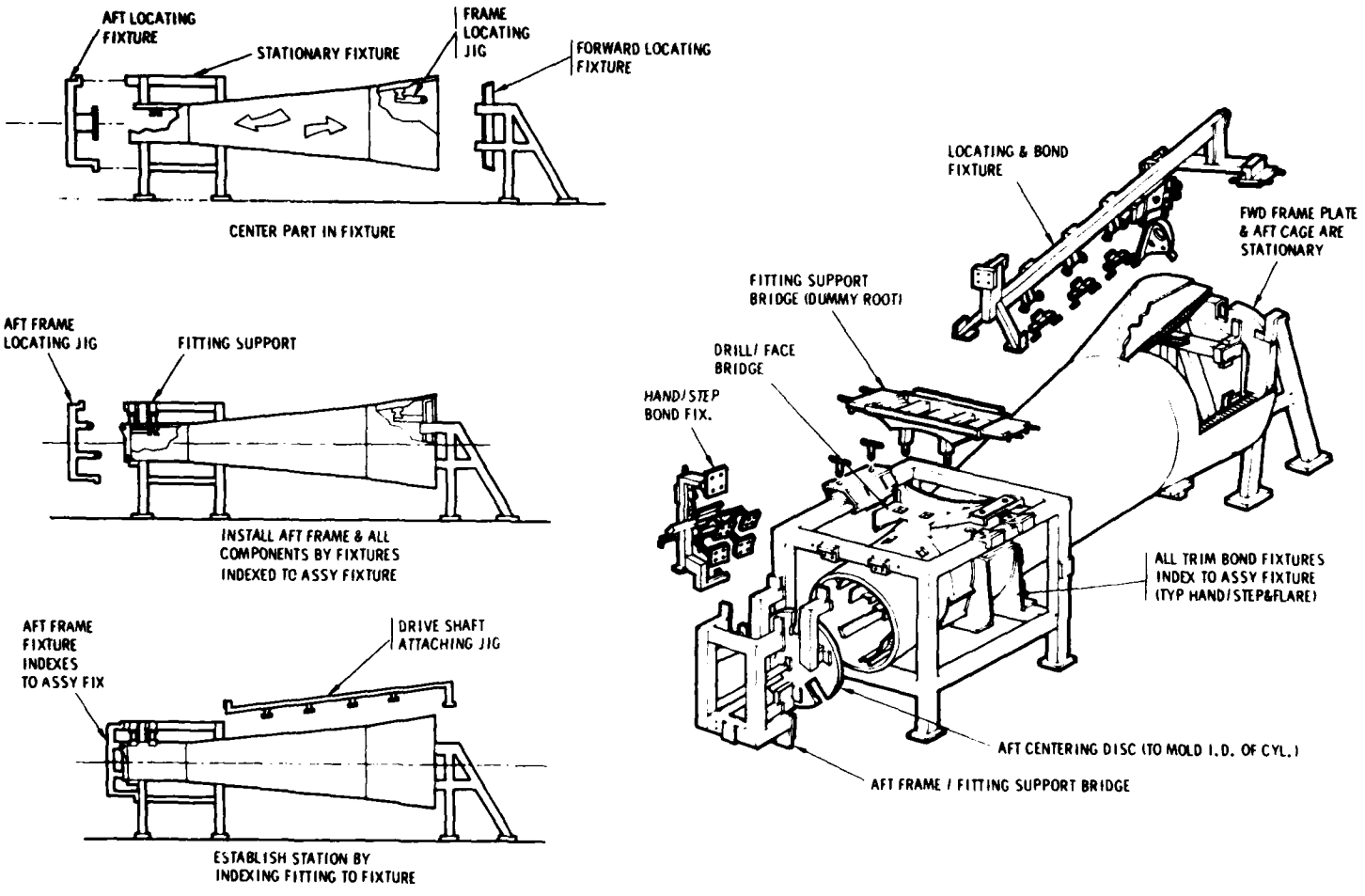


Figure 82. Final Assembly Fixture Functions

NON-DESTRUCTIVE EVALUATION (NDE)

NDE is planned to ensure that all CTS components meet the design criteria, that they are free of internal defects, and that they have the proper internal geometry. Reports of all NDE activities are maintained by component serial number for permanent record.

Types of defects that could degrade the components' performance are:

- Delaminations
- Unbonded areas
- Porosity or voids
- Resin rich or resin starved areas
- Geometry of internal spars
- Thick bond lines
- Foreign objects
- Defects in metal attach fittings

The importance of these defects varies with their location and size in components. Tentative criteria for various areas and sections of the CTS components are given below.

Figure 83 indicates critical zones A, B, and C of the tailboom. Allowable limits for disbonds and delaminations that may occur in the ring stiffener, sandwich wall of the shell mainframe, frames, intercostals, and fairing hinge supports are detailed in Table 11. NDE techniques scheduled for use on tailboom hardware are listed in Table 12.

Figure 84 delineates critical zones A and B of the vertical tail. Table 13 lists the limits for disbonds and delaminations allowed in the preassembly structure and in the completed item. Table 14 is a matrix of NDE techniques scheduled for vertical tail components.

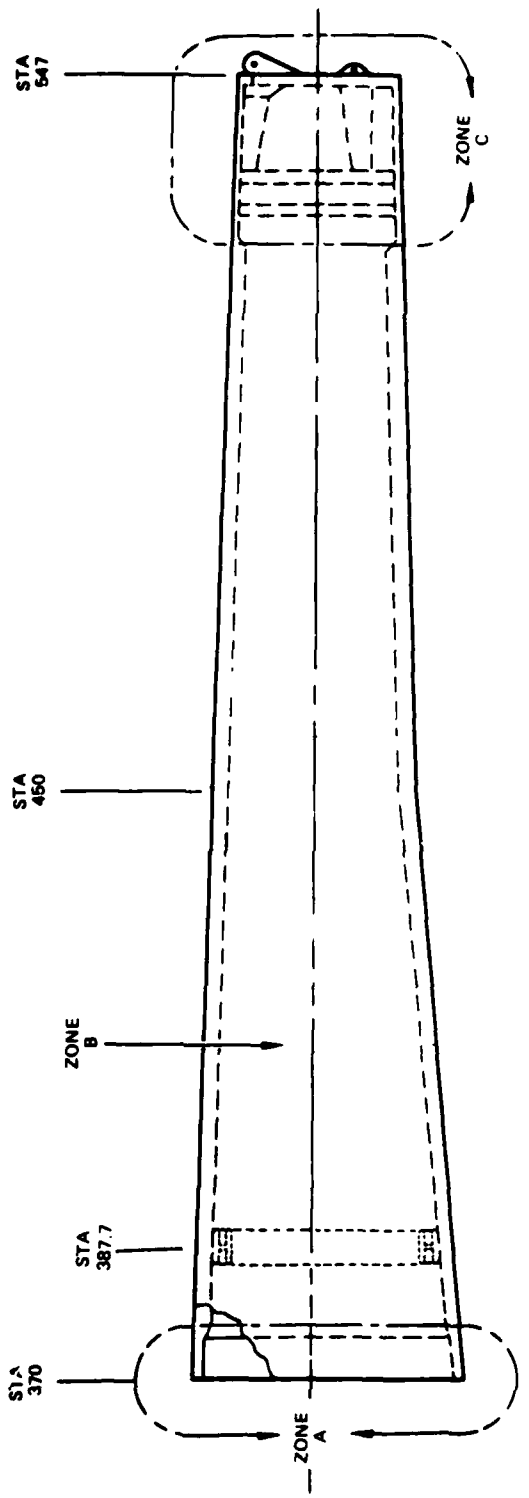


Figure 83. Composite Tailboom Inspection Zones

TABLE 11. TAILBOOM CRITICAL DISBOND AND DELAMINATION LIMITS

Zone A	
1. Solid Laminate:	No single delamination to be larger than 0.5 inch in diameter.
	No delaminated areas to be within 3.0 inches of each other.
	No more than 10 percent delamination allowed.
2. Sandwich Wall:	No single delamination or disbond to be larger than 1.0 inch in diameter.
	No delaminated or disbonded areas to be within 3.0 inches of each other.
	No more than 10 percent delamination or disbond allowed.
Zone B	
1. Sandwich Wall:	No single delamination or disbond to be larger than 1.0 inch in diameter.
	No delaminations or disbonded areas to be within 3.0 inches of each other.
	No more than 10 percent delamination of disbond allowed.
2. Fairing Hinge Supports:	No more than 20 percent disbond allowed.
	No single disbond to be more than 1.0 square inch.
	No two disbonded areas to be within 2.0 inches of each other.
Zone C	
1. Shell:	No single delamination to be larger than 1.0 inch in diameter.
	No delaminated areas to be within 3.0 inches of each other.
	No more than 1.0 percent delamination allowed.
2. Frame:	No single disbond to be more than 1.0 square inch.
	No two delaminated areas to be within 2.0 inches of each other.
	No more than 10 percent disbond allowed.
3. Intercostals:	No single disbond to be more than 1.0 square inch.
	No two delaminated areas to be within 2.0 inches of each other.
	No more than 10 percent disbond allowed.

TABLE 12. POTENTIAL NDE TECHNIQUES – TAILBOOM

Section	X-ray	Acoustic Emission (1)	Ultrasonic			Impulse Response	Visual (2)	Stiffness Test
			Harmonic Bond Tester	C-Scan	Bondascope			
Conical Section	X	X	X	X	X	X	X	X
Frames/ Intercostals			X		X	X	X	

(1) During static, dynamic and fatigue testing.

(2) Cut edges, exterior bond surfaces, etc.

TABLE 13. VERTICAL TAIL - CENTER BOX CRITICAL DISBOND AND DELAMINATION LIMITS

Zone A	
1. Laminate:	No delamination to be over 1.0 inch in diameter. No skin delamination to be within 3.0 inches of another. No more than 15 percent delamination allowed.
Zone B	
1. Preassembly Sandwich Structure:	No single void to be more than 0.5 inch in any direction, and not tangent to any edge. No two void areas to be within 3.0 inches of each other. No more than 10 percent delamination allowed.
2. Integrally Fabricated Attach Brackets:	No disbond to be over 0.5 inch in diameter. No two disbond areas to be within 3.0 inches of each other. No more than 10 percent disbond allowed.

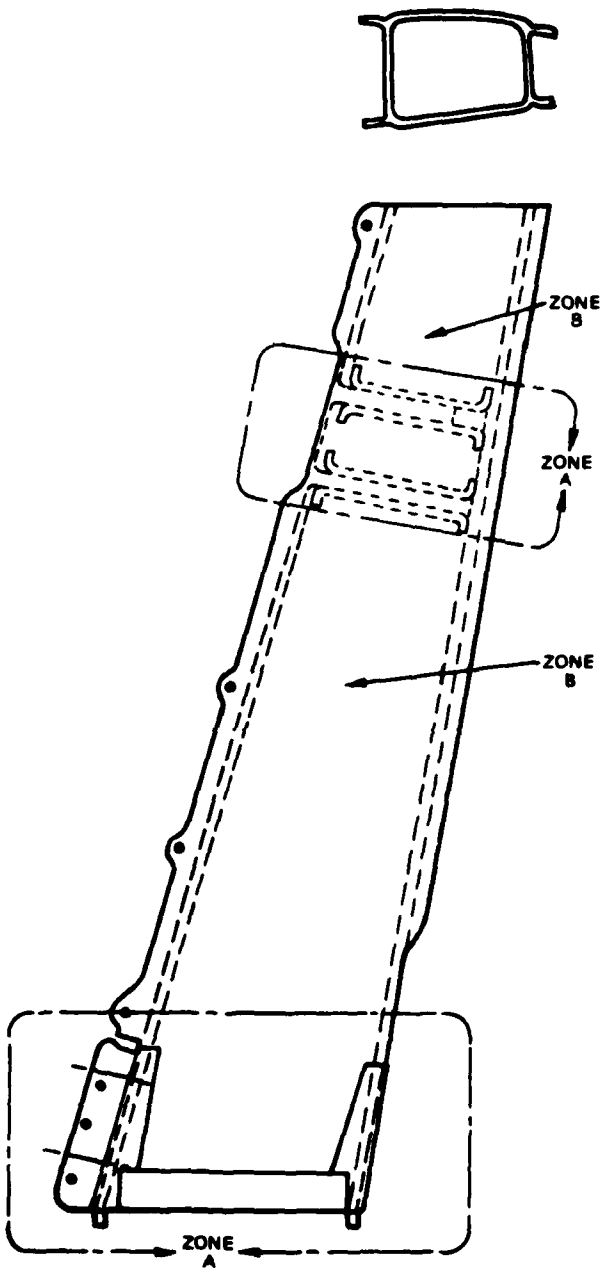


Figure 84. Composite Vertical Tail Inspection Zones

TABLE 14. POTENTIAL NDE TECHNIQUES – VERTICAL

Section	X-ray	Acoustic Emission (1)	Ultrasonic			Impulse Response	Visual (2)	Stiffness Test
			Harmonic Bond Tester	C-Scan	Bondascope			
Spar Box	X	X	X	X	X	X	X	X
Ribs			X		X		X	
T-Flanges			X		X		X	

(1) During static, dynamic, fatigue testing.

(2) Cut edges, exterior bond surfaces, etc.

The stabilators' critical zones A and B are defined in Figure 85. Tolerances for bond and laminate defects permitted in preassembly components, attachable details, and the completed structure are delineated in Table 15. Table 16 presents the NDE procedures to be used to evaluate the integrity of stabilator structural parts.

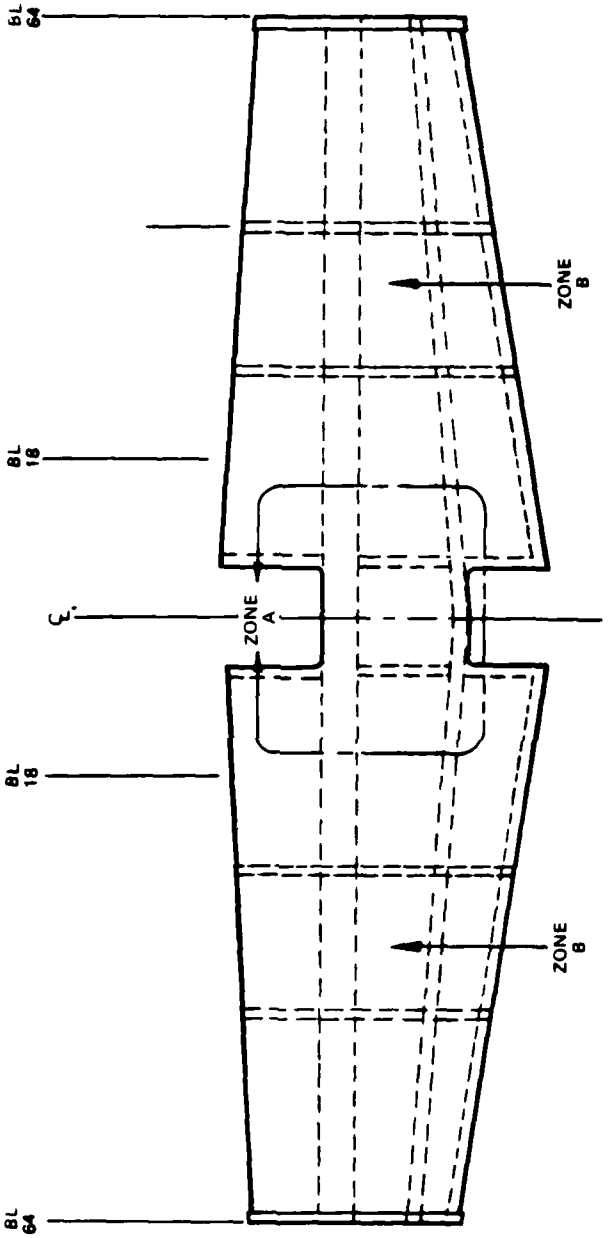


Figure 85. Composite Stabilator Inspection Zones

TABLE 15. STABILATOR CRITICAL DISBOND AND DELAMINATION LIMITS

Preassembly and Attachable Details	
1. Skin:	No single delamination to be over 1.0 square inch. No delamination to be within 3.0 inches of another. No more than 10 percent delamination allowed.
2. Forward and Aft Spar Assemblies:	No delamination to be over 1.0 square inch. No single delaminations to be within 3.0 inches of each other. No more than 10 percent delamination allowed.
3. Ribs and Caps:	No skin delamination to be over 1.0 square inch. No single delamination to be within 3.0 inches of another. No more than 10 percent delamination allowed.

TABLE 15. STABILATOR CRITICAL DISBOND AND DELAMINATION LIMITS (CONT)

Final Assembly	
Zone A	
1. Forward and Aft Spar Assembly Bonds:	No more than 10 percent disbond allowed. No disbond to be larger than 1.0 square inch. No disbonded area to be within 3.0 inches of another.
2. Ribs and Closeouts:	No more than 10 percent for ribs and 20 percent for closeout disbonds allowed along attach flanges. No area of disbond to extend across bonding flange.
Zone B	
1. Forward and Aft Spar Assembly Bond:	No more than 10 percent disbond allowed. No single disbond to be more than 1.0 square inch. No disbonded area to be within 3.0 inches of another.
2. Trailing Edge Bond Joint:	No more than 10 percent disbond allowed. No voids allowed 0.5 inch from trailing edge.
3. Ribs and Caps:	No more than 10 percent disbond allowed. No disbond to extend across bonding flange.

TABLE 16. POTENTIAL NDE TECHNIQUES – STABILATOR

Section	X-ray	Ultrasonic			Impulse Response	Visual (1)	Stiffness Test
		Harmonic Bond Tester	C-Scan	Bondascope			
Spars	X	X	X	X		X	
Ribs		X	X	X		X	
Assembly		X		X	X	X	X

(1) Cut edges, exterior bond surfaces, etc.

DESIGN TO UNIT PRODUCTION COST

The design to unit production cost (DTUPC) assessment (in Second Quarter 1981 dollars) is based on HHI's experience with procuring and fabricating composites and metal components, and assembling them. The estimate for fabricating composite components is divided into those that are labor-intensive and labor non-intensive. Different process improvement factors are used for each category. The procurement of the metal components is related to similar components in the prototype AH-64A helicopter. The initial estimates are made for the Serial Number (S/N) 010 unit. The costs are worked backward to the S/N-001 unit and then forward to the average for the 536 shipset production base.

The labor required to fabricate the three major composite components is estimated in Tables 17, 18, and 19 for each step in the process. The composite materials required for three CTS components are listed in Table 20.

The fabricated metal components that go into each of the three CTS components and connect them together are estimated in Table 21.

The production improvement factors used in this analysis are:

- Composite fabrication (labor intensive) 80 percent
- Composite fabrication (labor non-intensive) 85 percent
- Purchased metal components 95 percent
- Purchased composite materials 100 percent

The mathematical ratios for these factors relative to the first unit, tenth unit, and cumulative average for 536 units are:

Improvement Factor	S/N-001	S/N-010	Cumulative Average, 536
80 percent	1.0000	0.3286	0.1373
85 percent	1.0000	0.4636	0.2291
95 percent	1.0000	0.8193	0.6281
100 percent	1.0000	1.0000	1.0000

TABLE 17. LABOR FOR TAILBOOM (S/N-010)

		Manminutes	
		Labor Intensive	Labor Non-Intensive
Clean and prepare tailboom mandrel	2 men, 1 hour	120	-
Clean and prepare frame and intercostal molds	7 molds, 1 man per mold per 1/4 hour	105	-
Cut out fabric	2 men, 1/2 hour	60	-
Layup frames and intercostals	7 molds, 2 men per mold per 1/2 hour	420	-
Cure frames and intercostals	1 man, 6 hours	360	-
Trim and prepare honeycomb	2 men, 1 hour	120	-
Trim and assemble frames and intercostals	2 men, 2 hours	240	-
Assembly tailboom mandrel	2 men, 1/2 hour	60	-
Mount mandrel in winding machine	2 men, 1/4 hour	-	30
Wind inner skin	1 man, 50 lb, 8 lb/ hour	-	375
Place film adhesive	2 men, 1/4 hour	-	30
Place honeycomb	2 men, 1/2 hour	-	60

TABLE 17. LABOR FOR TAILBOOM (S/N-019) (CONT)

		Manminutes	
		Labor Intensive	Labor Non-Intensive
Place film adhesive	2 men, 1/4 hour	-	30
Place doublers and fillers	2 men, 1/2 hour	60	-
Wind outer skin	1 man, 50 lb, 8 lb/ hour	-	375
Install peel ply	2 men, 1/4 hour	-	30
Overwind	1 man, 1 ft/min, 20 ft	-	20
Phase I cure	1 man, 6 hours	-	360
Remove tailboom from mandrel	2 men, 3/4 hour	-	90
Install frames, etc.	2 men, 1 hour	120	-
Post cure	1 man, 6 hours	-	360
Trim, drill, install metal fittings, finish machine	3 men, 8 hours	1440	-
Inspect	1 man, 2 hours	-	120
Paint	1 man, 2 hours	-	120
Weigh	2 men, 1/4 hour	-	30
Pack for shipment	2 men, 1/4 hour	-	30
	TOUCH TIME	3105	2060

TABLE 18. LABOR FOR STABILATOR (S/N-010)

		Manminutes	
		Labor Intensive	Labor Non-Intensive
Clean and prepare skin and broadgoods mandrels	2 men, 1 hour	120	-
Clean and prepare rib, spar and skin molds	32 molds, 1 man, 1/4 hour	480	-
Wind broadgoods	1 man, 40 lb, 8 lb/ hour	-	300
Cut out fabric and broadgoods	2 men, 2 hours	240	-
Lay up ribs and spars	16 molds, 1 man/ mold, 1/2 hr	480	-
Wind skins	1 man, 30 lb, 8 lb/ hour	-	225
Cut out skins	2 men, 1/2 hour	60	-
Lay up skins	2 men, 1 hour	120	-
Phase I cure ribs, spars and skins	(A)	-	0
Trim ribs, spars and skins	2 men, 3 hours	-	960

(A) Coincide with tailboom and vertical fin

TABLE 18. LABOR FOR STABILATOR (S/N-010) (CONT)

		Manminutes	
		Labor Intensive	Labor Non-Intensive
Assembly ribs and spars	2 men, 2 hours	-	240
Cure rib/spar assembly	(A)	-	0
Assemble skins	2 men, 1/2 hour	60	-
Cure stabilator	(A)	-	0
Trim, drill, install metal fittings	2 men, 3 hours	960	-
Inspect	1 man, 1 hour	-	60
Paint	1 man, 1 hour	-	60
Weigh	2 men, 1/8 hour	-	15
Pack for shipment	1 man, 1/4 hour	-	15
	TOUCH TIME	2520	1875

(A) Coincide with tailboom and vertical fin

TABLE 19. LABOR FOR VERTICAL TAIL (S/N-010)

		Manminutes	
		Labor Intensive	Labor Non-Intensive
Clean and prepare spar mandrel	1 man, 1 hour	60	-
Clean and prepare bulkhead and angle molds	8 molds, 1 man, 1/4 hour	120	-
Cut out fabric	1 man, 1 hour	60	-
Lay up bulkheads and angles	8 molds, 2 men/ mold, 1/2 hour	480	-
Cure bulkheads	(B)	-	0
Trim bulkheads and prepare bulkheads and angles	2 men, 2 hours	240	-
Assemble fin spar mandrel	2 men, 1 hour	120	-
Wind inner skin	1 man, 10 lb, 8 lb/ hour	-	75
Place film adhesive	2 men, 1/8 hour	-	15
Place honeycomb	2 men, 1/4 hour	-	30
Place film adhesive	2 men, 1/8 hour	-	15

(B) Coincide with tailboom and stabilator

TABLE 19. LABOR FOR VERTICAL TAIL (S/N-010) (CONT)

		Manminutes	
		Labor Intensive	Labor Non-Intensive
Wind outer skin	1 man, 10 lb, 8 lb/ hour	-	75
Place peel ply	2 men, 1/8 hour	15	-
Place caul sheets	2 men, 1/4 hour	-	-
Overwind	1 man, 1 ft/min, 9 ft	-	10
Phase I Cure	(B)	-	0
Remove spar from mandrel	2 men, 1/4 hour	30	-
Install bulkheads and angles	2 men, hour	240	-
Post cure	(B)		
Trim, drill, install metal fittings, and finish machine	2 men, hours	480	-
Inspect	1 man, 1 hour	-	60
Paint	1 man, 1 hour	-	60

(B) Coincide with tailboom and stabilator

TABLE 19. LABOR FOR VERTICAL TAIL (S/N-010) (CONT)

		Manminutes	
		Labor Intensive	Labor Non-Intensive
Install leading and trailing edge fairings	2 men, 1/2 hour each	120	-
Weigh	2 men, 1/8 hour	-	15
Pack for shipping	1 man, 1/4 hour	-	15
	TOUCH TIME	1995	370

TABLE 20. COMPOSITE MATERIAL FOR CTS (SECOND QUARTER 1981 DOLLARS)

	Pounds	Square Feet	$ per Pound	$ per Square Foot	$
TAILBOOM					
Graphite	34.1		26.00		887
Kevlar	27.4		9.00		245
Resin	61.0		1.20		73
Honeycomb	13.0		150.00		1950
Film Adhesive		105		1.30	137
Miscellaneous Process Materials					150
25% Allowance for Scrap, Turnaround, etc.					861
				TOTAL	$4303
STABILATOR					
Graphite	11.3		26.00		294
Kevlar	7.0		9.00		63
Resin	18.0		1.20		22
Urethane	3.8		15.00		57
Film Adhesive		15.0		1.30	20
Miscellaneous Process Materials					100
25% Allowance for Scrap, Turnaround, etc.					139
				TOTAL	$ 695
VERTICAL TAIL					
Graphite	16.0		26.00		416
Resin	16.0		1.20		19
Honeycomb	3.0		150.00		450
Film Adhesive		40		1.30	52
Miscellaneous Process Materials					100
25% Allowance for Scrap, Turnaround, etc.					259
				TOTAL	$1296
				GRAND TOTAL	$6294

TABLE 21. COMPONENTS OTHER THAN STRUCTURAL COMPOSITES (SECOND QUARTER 1981 DOLLARS)

Tailboom		
One Sta 547 Frame	Based on $3800 Phase II frame + 40 percent inflation + added complexity and HHI material + machining estimate)	$ 5,700
Two Sta 532 Fittings	Based on size and complexity relative to 547 frame	700
Four Intercostal Clips	Based on size and complexity relative to 547 frame	400
One Jack Fitting	Based on $700 Phase II titanium fitting redesigned into stainless steel	400
One Set of Tail Rotor Shaft Supports	Phase II quote + 40 percent inflation	1,400
One Set of Maintenance Steps	HHI estimate	600
Miscellaneous Hardware, Fasteners, etc.	HHI estimate	1,200
	TOTAL	$10,400
Stabilator		
Two Stabilizer Hinge Fittings	Based on size and complexity relative to 547 frame	$ 1,000
Six Bathtub Fittings	Based on size and complexity relative to 547 frame	900
Two Actuator Fittings	Based on size and complexity relative to 547 frame	300
Miscellaneous Hardware, Fasteners, etc.		200
	TOTAL	$ 2,400
Vertical Tail		
Four Fin Attach Fittings	Based on size and complexity relative to 547 frame	$ 3,200
Nine Gearbox Attach Fittings	Based on size and complexity relative to 547 frame	2,000
Leading Edge, Trailing Edge, and Tip Fairings	Phase II quote	5,000
Four Maintenance Steps	HHI estimate	300
Miscellaneous Hardware, Fasteners, etc.	HHI estimate	800
	TOTAL	$11,300
	GRAND TOTAL	$22,800

A burdened labor rate of $35 per hour is assumed for composites fabrication and final assembly.

The DTUPC for the CTS is calculated in Table 22 and is summarized in Table 23. At the time that the CTS program was terminated, the AH-64A DTUPC for the comparable metal components was quoted by Teledyne Ryan Aeronautical (the AH-64A airframe subcontractor) as shown in Table 23. (Both composite and metal DTUPC's are rounded to the nearest $500.)

TABLE 22. DESIGN TO UNIT PRODUCTION COST
(SECOND QUARTER 1981 DOLLARS)

Tailboom

S/N-010 Labor Intensive + Support = 1-2/3* x 3105 = 5175 mm

Labor Non-Intensive + Support =

1-2/3* x 2060 = 3433 mm

Metal, etc. = $9100

Composites = $4303

S/N-001 Labor Intensive + Support =

$$\frac{5175}{60 \times 0.3286} = 262 \text{ mh}$$

Labor Non-Intensive + Support =

$$\frac{3433}{60 \times 0.4636} = \frac{123}{385 \text{ mh}}$$

Metal, etc. = $\frac{\$10,400}{0.8193} = \$12,694$

Composite = $\frac{\$4303}{1.0000} = \$4,303$

*2/3 factor for nonproductive personnel time

TABLE 22. DESIGN TO UNIT PRODUCTION COST (CONT)
(SECOND QUARTER 1981 DOLLARS)

Tailboom (Cont)

Cumulative Avg for 536	Labor Intensive	=	262 x 0.1323 x $35	=	$ 1,213
	Labor Non-Intensive	=	123 x 0.2291 x $35	=	986
	Metal	=	$12,694 x 0.6281	=	7,973
	Composite	=	$4,303 x 1.0000	=	4,303
			TOTAL		$14,475

Stabilator

S/N-010

Labor Intensive + Support = 1-2/3* x 2520 = 4200 mm

Labor Non-Intensive + Support =

1-2/3* x 1875 = 3125 mm

Metal, etc. = $2400

Composites = $695

*2/3 factor for nonproductive personnel time

TABLE 22. DESIGN TO UNIT PRODUCTION COST (CONT)
(SECOND QUARTER 1981 DOLLARS)

Stabilator (Cont)

S/N-001 Labor Intensive + Support =

$$\frac{4200}{60 \times 0.3286} = 213 \text{ mh}$$

Labor Non-Intensive + Support =

$$\frac{3125}{60 \times 0.4636} = \frac{112}{325 \text{ mh}}$$

Metal, etc. $= \frac{\$2400}{0.8193} = \$2929$

Composite $= \frac{\$695}{1.0000} \approx \$695$

Cumulative Avg for 536	Labor Intensive	=	213 x 0.1323 x $35	=	$ 986
	Labor Non-Intensive	=	112 x 0.2291 x $35	=	898
	Metal	=	$2929 x 0.6281	=	1,840
	Composite	=	$695 x 1.0000	=	695
			TOTAL		$ 4,419

TABLE 22. DESIGN TO UNIT PRODUCTION COST (CONT)
(SECOND QUARTER 1981 DOLLARS)

Vertical Tail

S/N-010 Labor Intensive + Support = 1-2/3* x 1995 = 3325 mm

Labor Non-Intensive + Support =

1-2/3* x 370 = 617 mm

Metal, etc. = $11,300

Composites = $1296

S/N-001 Labor Intensive + Support =

$$\frac{3575}{60 \times 0.3286} = 168 \text{ mh}$$

Labor Non-Intensive + Support =

$$\frac{617}{60 \times 0.4636} = \frac{22}{190 \text{ mh}}$$

Metal, etc. = $\frac{\$11,300}{0.8193}$ = $13,792

Composite = $\frac{\$1296}{1.0000}$ = $1296

*2/3 factor for nonproductive personnel time

TABLE 22. DESIGN TO UNIT PRODUCTION COST (CONT)
(SECOND QUARTER 1981 DOLLARS)

Vertical Tail (Cont)					
Cumulative Avg for 536	Labor Intensive	=	168 x 0.1323 x $35	=	$ 778
	Labor Non-Intensive	=	22 x 0.2291 x $35	=	176
	Metal	=	$13,792 x 0.6281	=	8,663
	Composite	=	$1296 x 1.0000	=	1,296
			TOTAL		$10,913

TABLE 23. DTUPC SUMMARY
(SECOND QUARTER
1981 DOLLARS)

Component	Metal	Composite	Saving
Tailboom	28,000	14,500	13,500
Stabilator	13,500	4,500	9,000
Vertical Tail	32,000	11,000	21,000
Airframe Modification to Accommodate CTS	-	2,500	<2,500>
	73,500	32,500	41,000

LABORATORY AND FLIGHT TEST PLANS

The CTS program did not progress far enough to conduct any structural testing. The plan was to conduct static, dynamic, fatigue, ballistic, and local impact tests to a level that would demonstrate safety of flight for a 25-hour flight test to be conducted under the MM&T program. When this was demonstrated to be successful, full airworthiness qualification in the laboratory and in flight would have been conducted.

LABORATORY TESTS

The CTS laboratory test article consists of the main structural elements of the CTS assembly including the tailboom, vertical tail spar, and stabilator. Fairings, etc., that have no influence on strength and/or stiffness are deleted from the test article.

Criteria for the laboratory tests include:

Static	The factor of safety is 1.5 for all conditions, with negligible permanent set at limit load, and no failure at ultimate load. The minimum margin of safety for all flight and ground loading conditions is zero for all structures, but in such cases where the analytical margin of safety is less than zero, a positive margin is to be demonstrated by static test.
Dynamic	The natural frequency of the CTS when installed on the AH-64A fuselage is such that the modal separation from the excitation frequencies is better than or equal to that of the metal tailboom and empennage to reduce CTS loads and vibration.
Fatigue	The CTS is designed to require no major overhaul in less than 4500 flight hours. Prior to determination of oscillatory load levels from flight strain surveys, weighted fatigue level of 10 percent of limit load for the tailboom and 30 percent for the stabilator and vertical tail is used for design purposes. The distribution of maneuver load cycles used in the fatigue analysis calculations is derived from the flight profile shown in Table 24.

Ballistic	The CTS ballistic survivability test consists of impacting the structure with a 23mm HEI-T projectile while the CTS is under load, and then applying static loading to destruction.
Impact	After static test failure, a local impact test that simulates anticipated service abuse will be conducted on the tailboom and stabilator. Type, location, and extent of damage will be recorded.

The test article shown in Figure 86 is mounted at Station 370 on a steel fixture. Straps, loading frames, or whiffletrees are attached to the CTS for introducing loads. Hydraulic loading jacks apply the test loads between the airframe attachments and the support framework. The tare weight of the attachments, linkages, and dummy fixtures is counterbalanced by hydraulic compensation in the loading system as well as by counterweights, pulleys, and cable.

Instrumentation includes strain and rosette gages mounted as shown in Figures 87 through 92. The same gages and locations are used in the laboratory test and in flight. Prior to test and at specified times during the tests, all gages are calibrated against standards.

Predicted loading conditions are established based on AH-64A flight tests, and modes of failure are predicted for comparison with actual test results.

FLIGHT TESTS

Limited flight tests are conducted on the load survey air vehicle for a total of 25 hours and to a level of 80 percent of the AH-64A flight envelope. Flight conditions are summarized in Table 24.

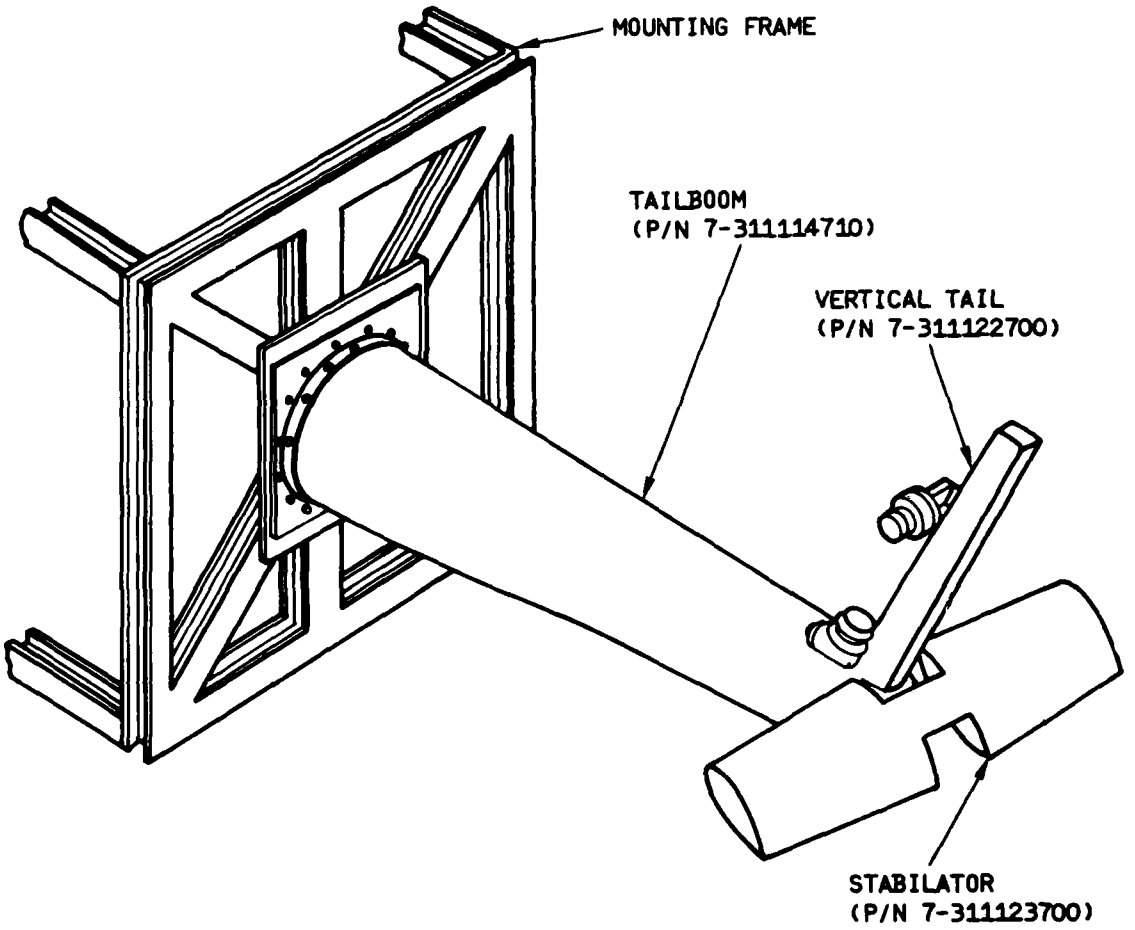


Figure 86. Schematic Static/Fatigue Test Setup

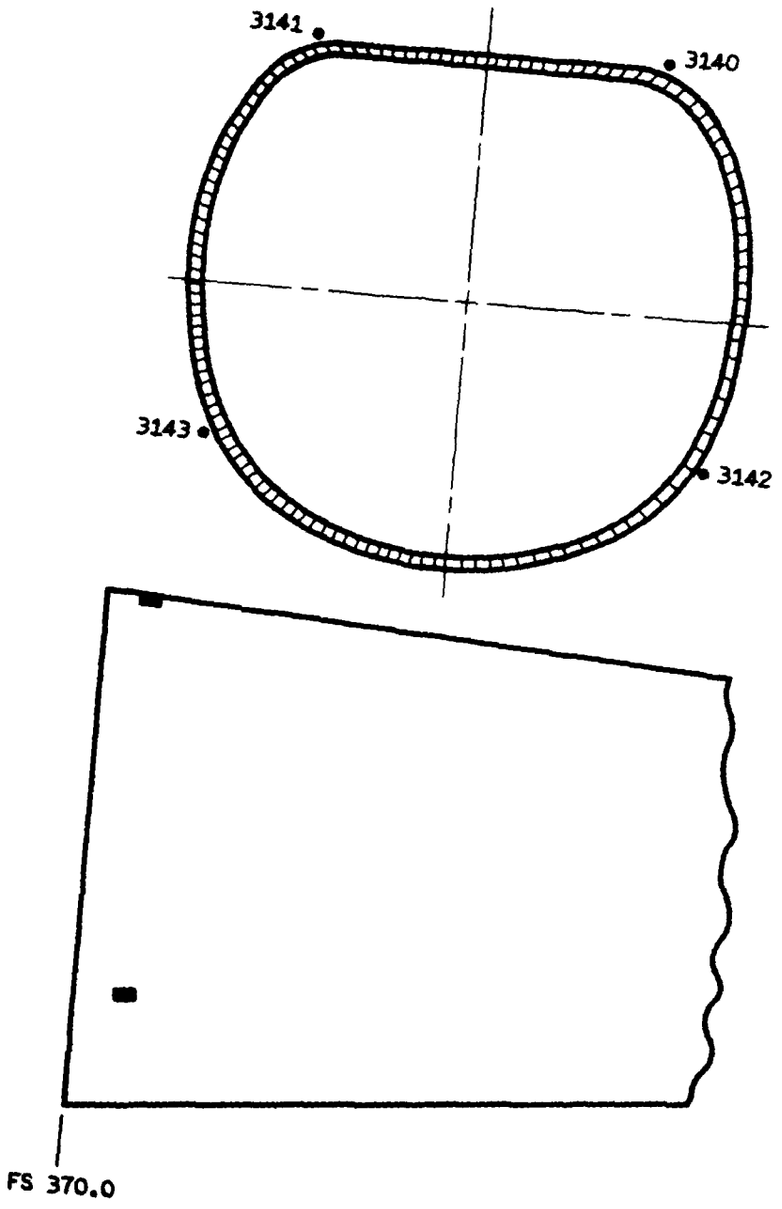


Figure 87. Strain Gage Installation FS 375

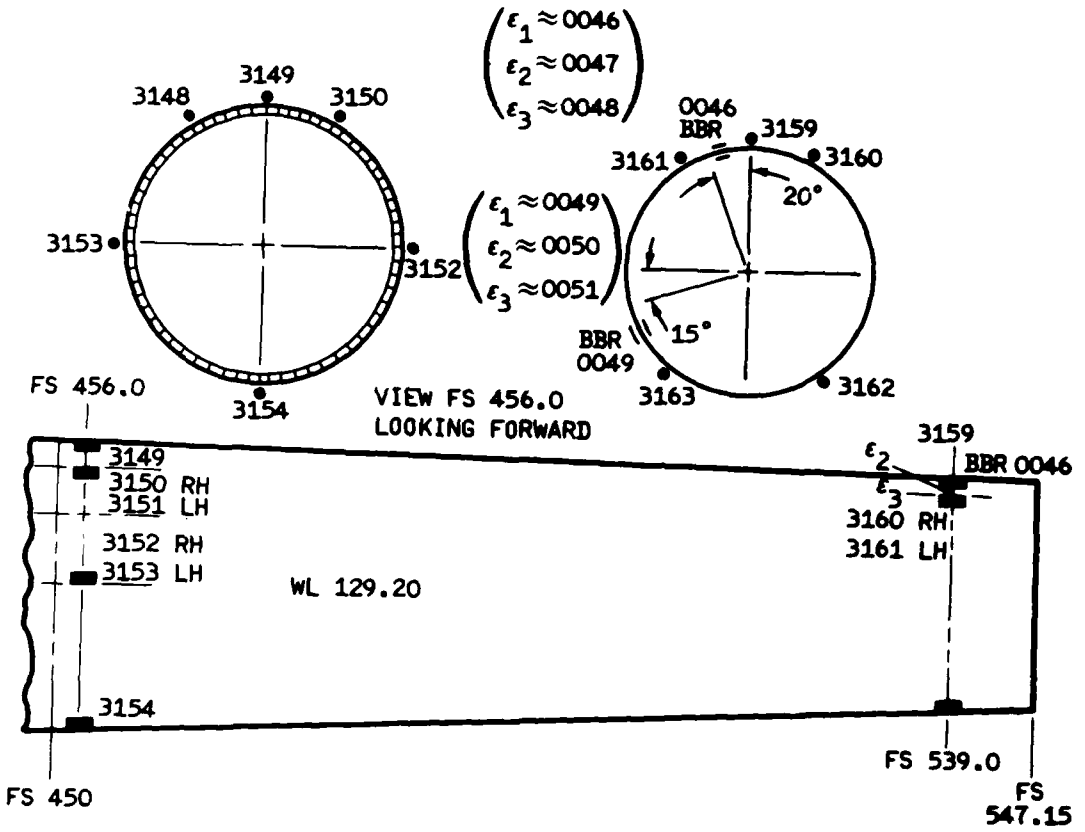


Figure 88. Strain Gage Installation - Tail Boom FS 539

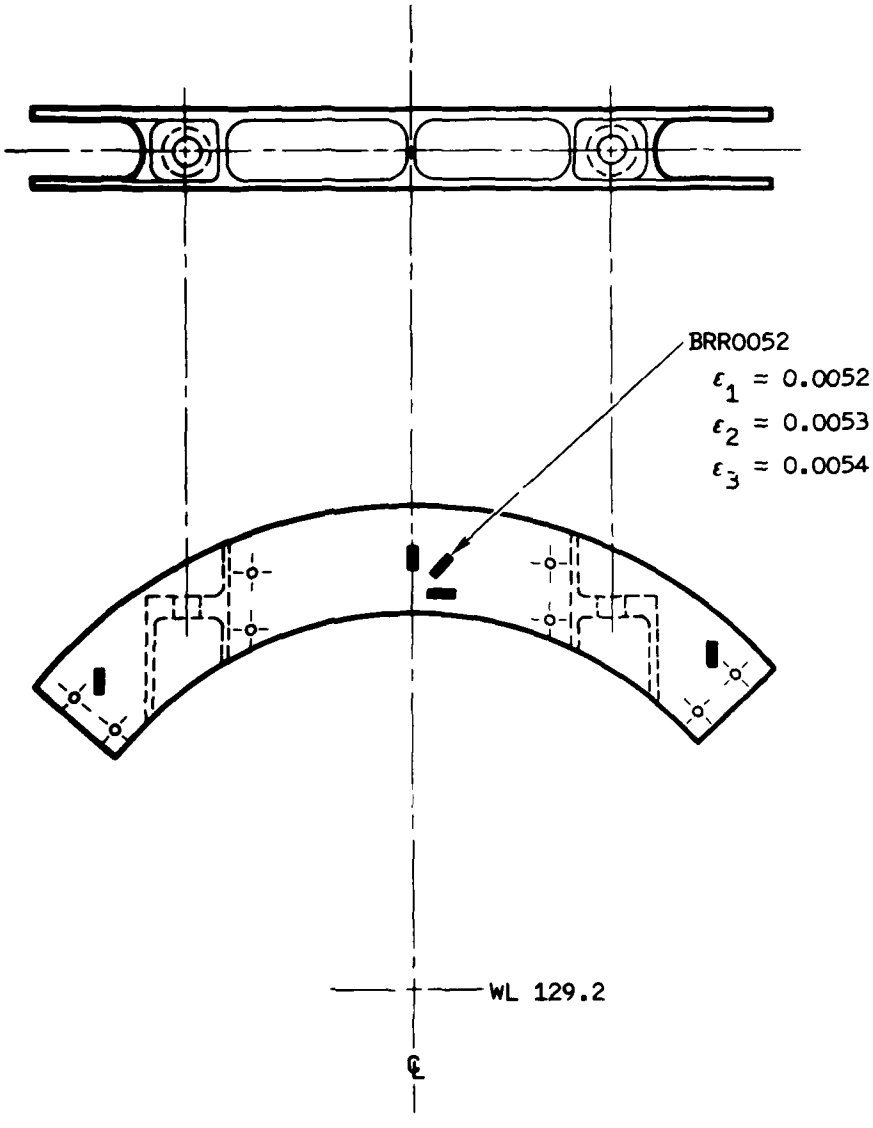


Figure 89. Strain Gage Installation - Vertical Stabilizer Fitting FS 530.09

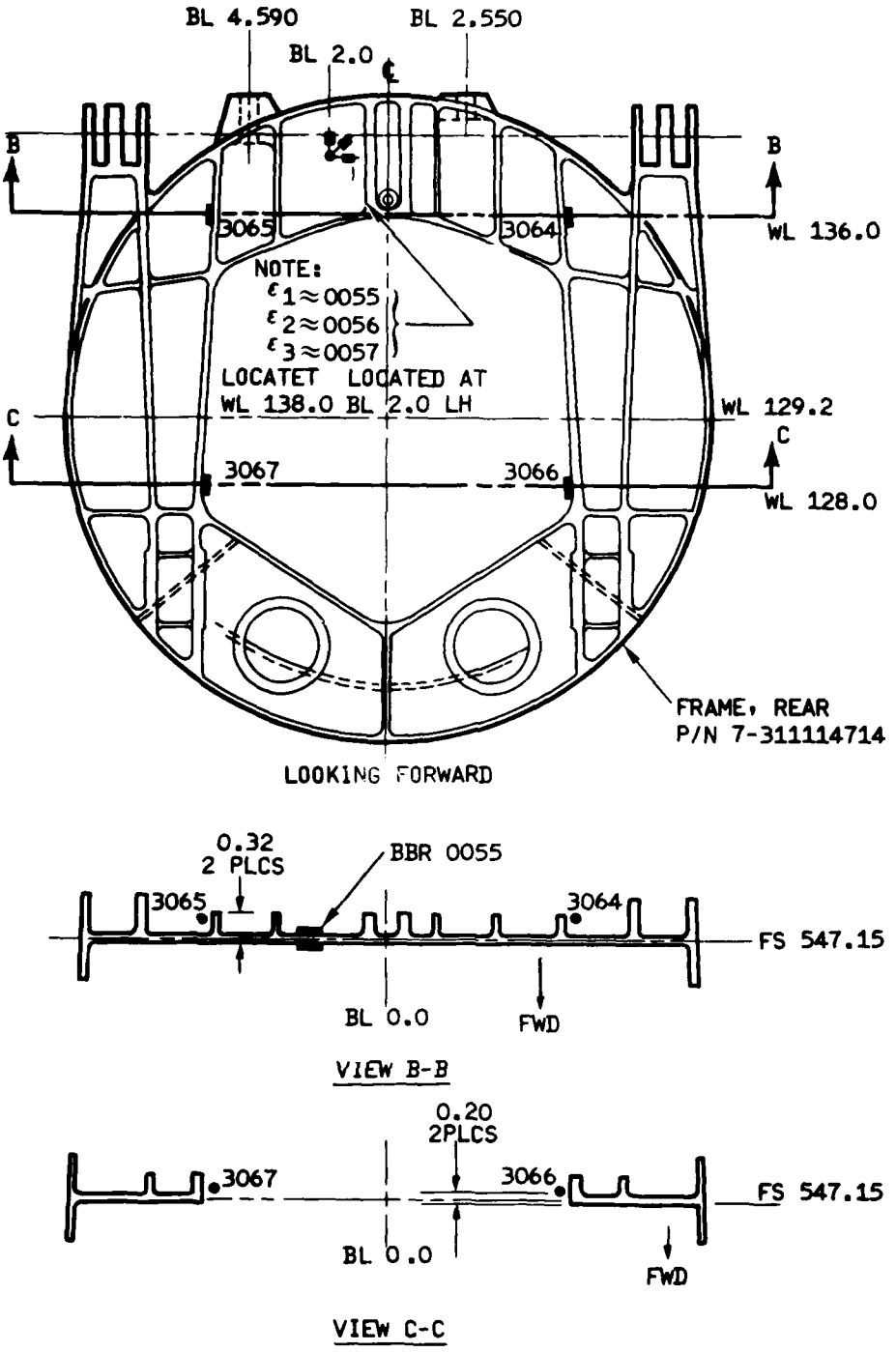


Figure 90. Strain Gage Installation - Rear Frame FS 547

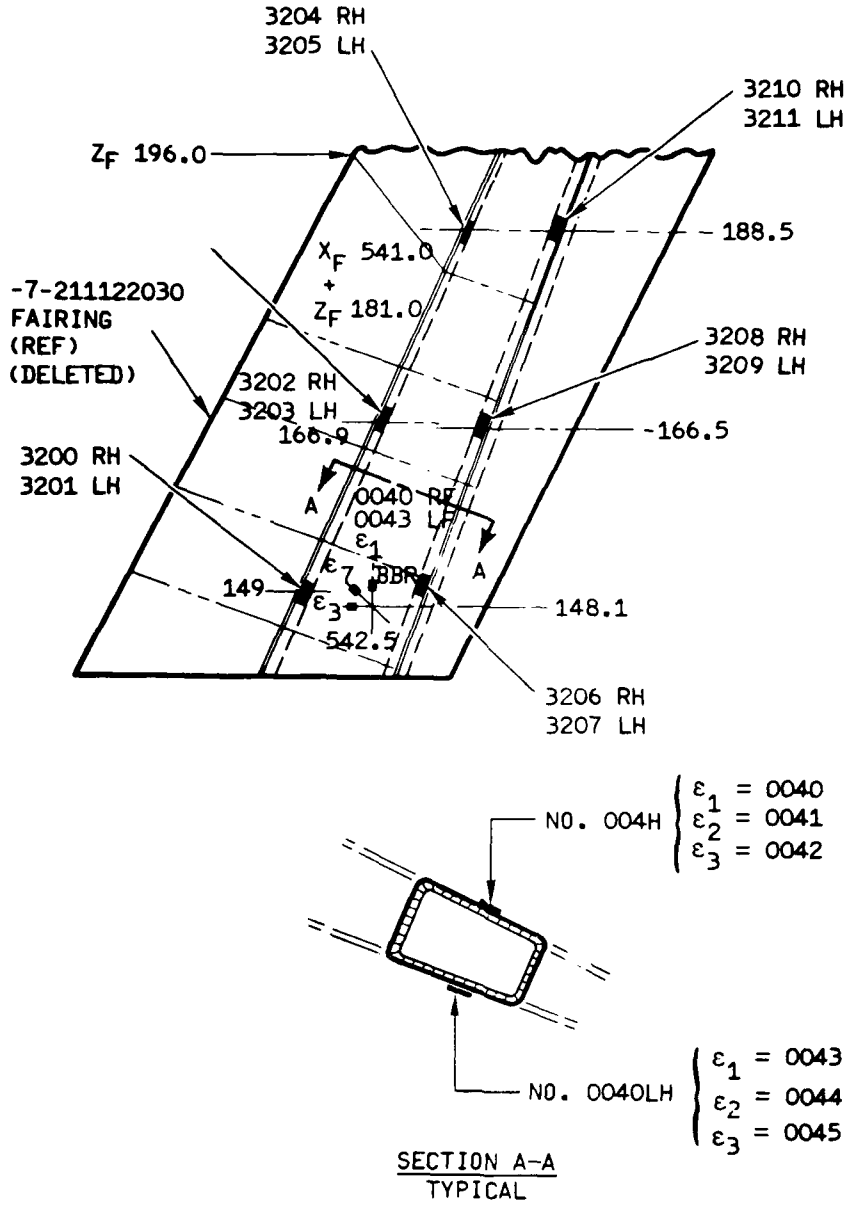


Figure 91. Strain Gage Installation - Vertical Stabilizer Spar

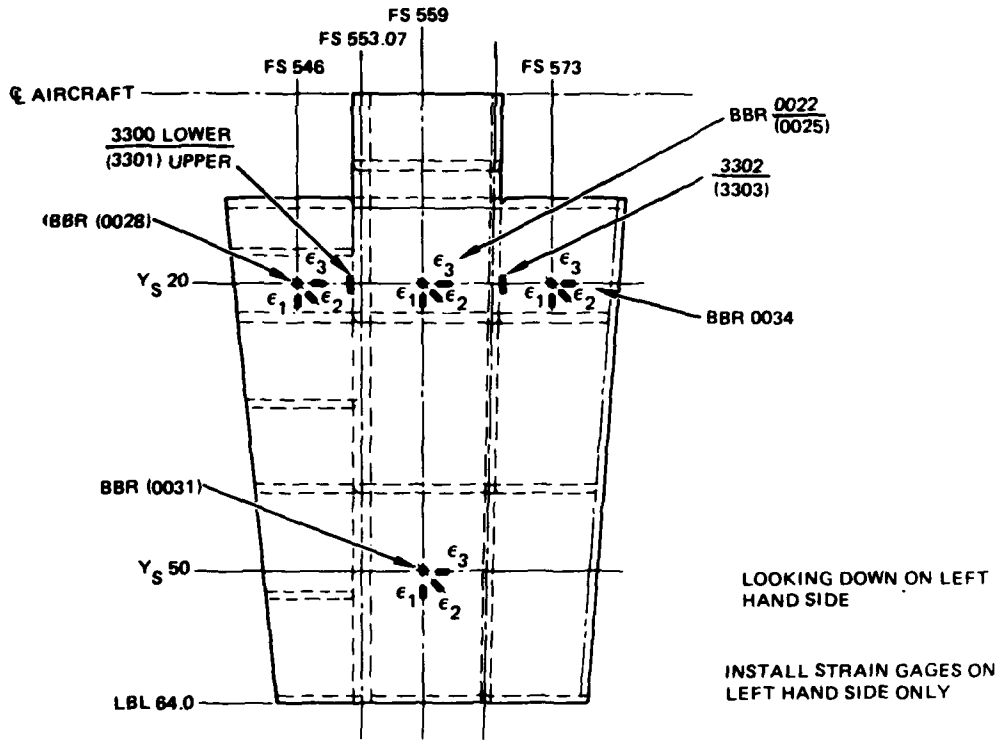


FIGURE 107. STRAIN GAGE INSTALLATION – STABILATOR SPAR 4 AXIAL STRAIN GAGE, SKIN 5 BBRs

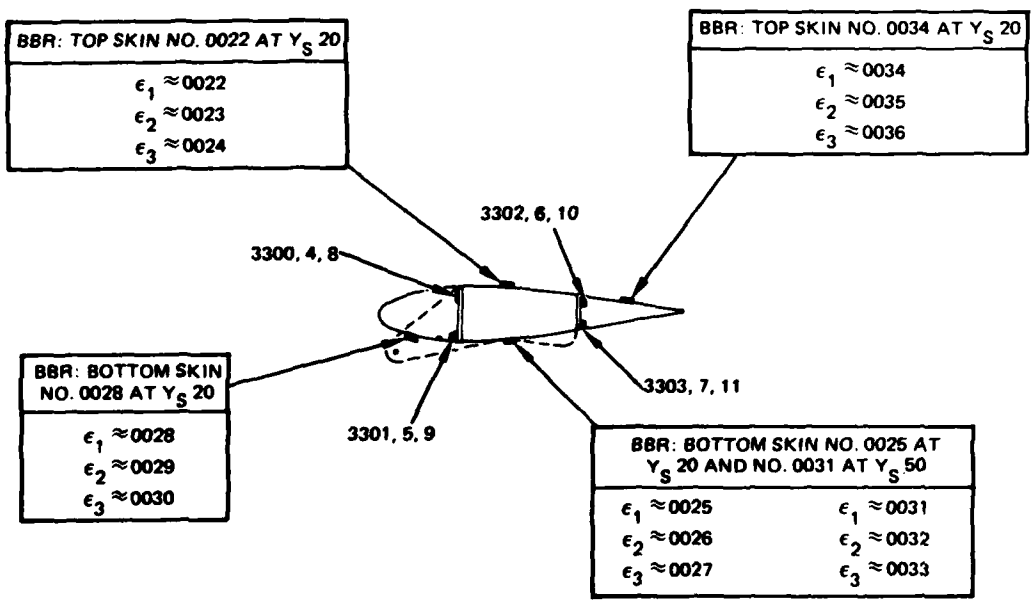


Figure 92. Strain Gage Installation - Stabilator Spar

TABLE 24. FLIGHT PROFILE

Speed Spectrum		Maneuver Spectrum		
Speed	Speed Envelope % Life	Peak Nz at cg	Time at g (sec)	Cumulative Exceedances per 4500 hours
Hover	23.0	3.00	0.8	200
(HOGE)		2.75	1.2	500
0.5 V_H	3.0	2.50	1.7	1,000
0.6 V_H	6.0	2.25	2.8	2,000
0.7 V_H	15.0	2.00	4.0	5,000
0.8 V_H	15.0	1.75	6.0	10,000
0.9 V_H	15.0	1.50	10.0	20,000
1.0 V_H	10.0	1.25	12.0	150,000
1.2 V_H	3.0			
Sideward Flight	7.0	0.75	4.2	60,000
Rearward Flight		0.50	2.8	8,000
Vertical Flight		0.25	2.5	1,000
Ground	3.0	0	2.0	200
Total	100.0			

NOTES:

1. 60 percent of maneuvers shall be unsymmetrical maneuvers.
   40 percent of maneuvers shall be symmetrical maneuvers.
2. Climb and descent time shall be distributed as follows:
   5 percent of level flight/moderate maneuvering time in ascending mode.
   8 percent of level flight/moderate maneuvering time in descending mode.
3. Gross weight shall be distributed as follows:
   40 percent of life at basic structural design gross weight.
   40 percent of life at basic structural design gross weight minus payload and 50 percent fuel.
   20 percent of life at maximum alternate gross weight.
4. The average time per maneuver considering all types of maneuvers shall be in accordance with the time at g column.
5. Life shall be divided approximately as follows:
   70 percent level flight with moderate maneuvering (0.75-1.25 g).
   20 percent pull up and turn maneuvers.
   10 percent other maneuvers (control reversals, auto rotation, accelerations/decelerations).
6. Firing loads shall be additive to the above profile. Assume 810 firing anti-armor missions with 55 percent of mission ordnance (MO) expended during HOGE. Assume 875 firing escort or covering missions with 20 percent of MO expended during HOGE. Distribution of remaining MO should be in accordance with the forward flight spectrum.
7. The altitude spectrum shall be:
   80 percent at sea level-STD (representing low altitude flight).
   20 percent at 4000 ft 95°F (representing high altitude flight).
8. GAG cycles: Three engine start/stop cycles per flight hour.

IMPLICATIONS FOR PRODUCTION

The CTS is designed to be a lightweight, low cost, integrated replacement for the AH-64A's metal tail section. As designed, it is attached to the forward fuselage structure at Sta 370 by a double row of HiLok fasteners and requires a modification of the metal airframe at this point. The joint thus formed imposes a small penalty that is well worth accepting in light of the overall benefits of the CTS that are a 71 pound weight saving (including a 7.6 pound weight penalty for the joint) and a $41,000 production cost saving (including a $2,500 cost penalty for the joint).

This is the most efficient CTS that can be used, but just before the design work was terminated, a plan was proposed to change to a "mix-and-match" configuration. In this arrangement, any or all of the three composite components could be used interchangeably with the AH-64A metal components. This work was not pursued beyond a preliminary estimate of the design work required and the potential impacts of such a configuration. The unique, integrated design of the CTS that came out of the program is the most effective configuration from a cost and weight standpoint, while the "mix-and-match" design would be easier to introduce into the AAH production line. Table 25 describes the CTS changes needed to accomplish "mix-and-match." Figures 93 and 94 compare the CTS as designed with the two most probable "mix-and-match" arrangements:

- Metal tailboom with composite stabilator and vertical tail
- Composite tailboom, stabilator, and vertical tail

with the basic metal configuration in terms of weight and cost. In addition to weighing more and costing more, the "mix-and-match" design would have the following consequences:

- Reduced vertical tail incidence increases CFTR flapping and potentially reduces its fatigue life
- Reduced stabilator hinge space increases hinge loads and reduces hinge life

TABLE 25. CTS DESIGN CHANGES TO ACHIEVE "MIX-AND-MATCH"

Vertical Tail

- Reduce incidence to match metal vertical tail
- Locate stabilator hinge fittings on vertical tail spar instead of on tailboom aft bulkhead
- Change root fittings and bolt pattern for attachment to tailboom

Stabilator

- Modify hinge fittings to match fittings on vertical tail spar
- Modify spar/rib structure to accommodate new hinge fitting location

Tailboom

- Change tailboom aft structure and bolt pattern for attaching vertical tail

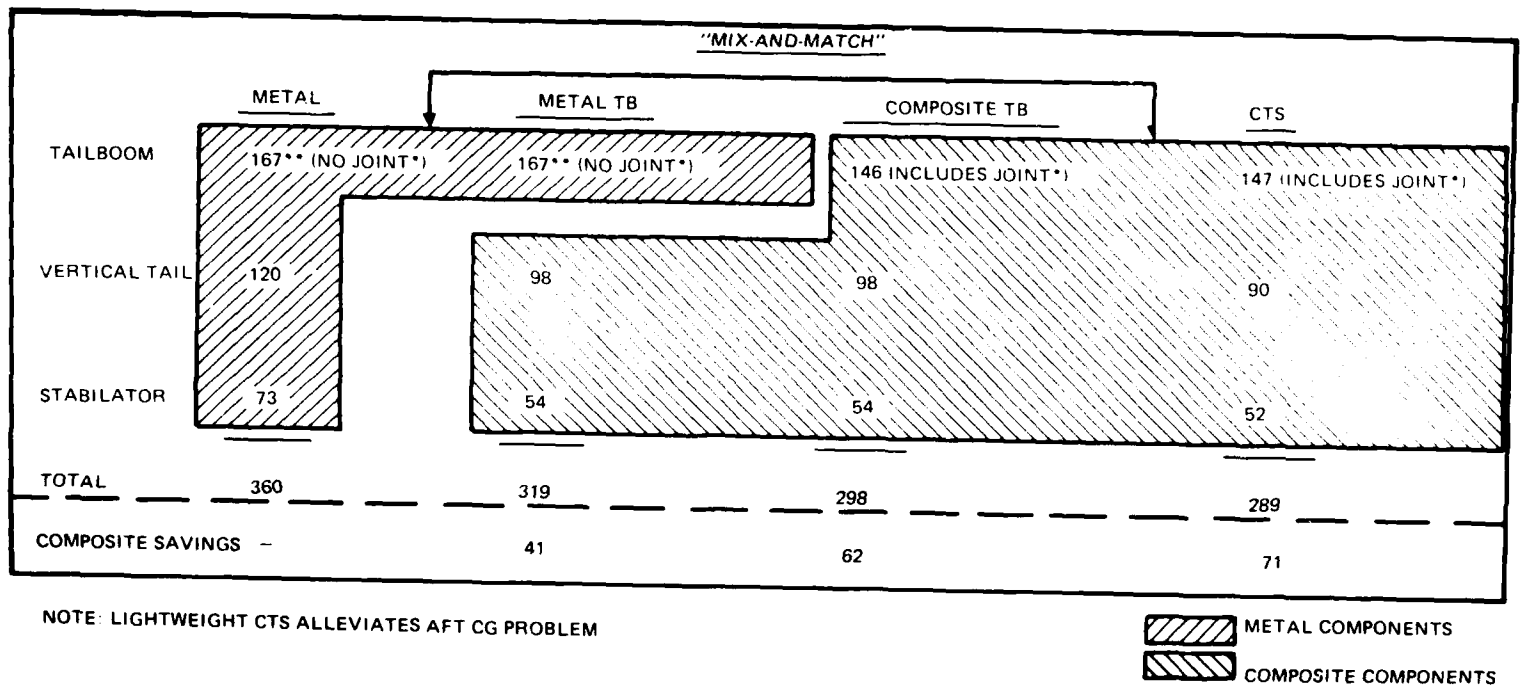


Figure 93. Tail Section Weight Comparison

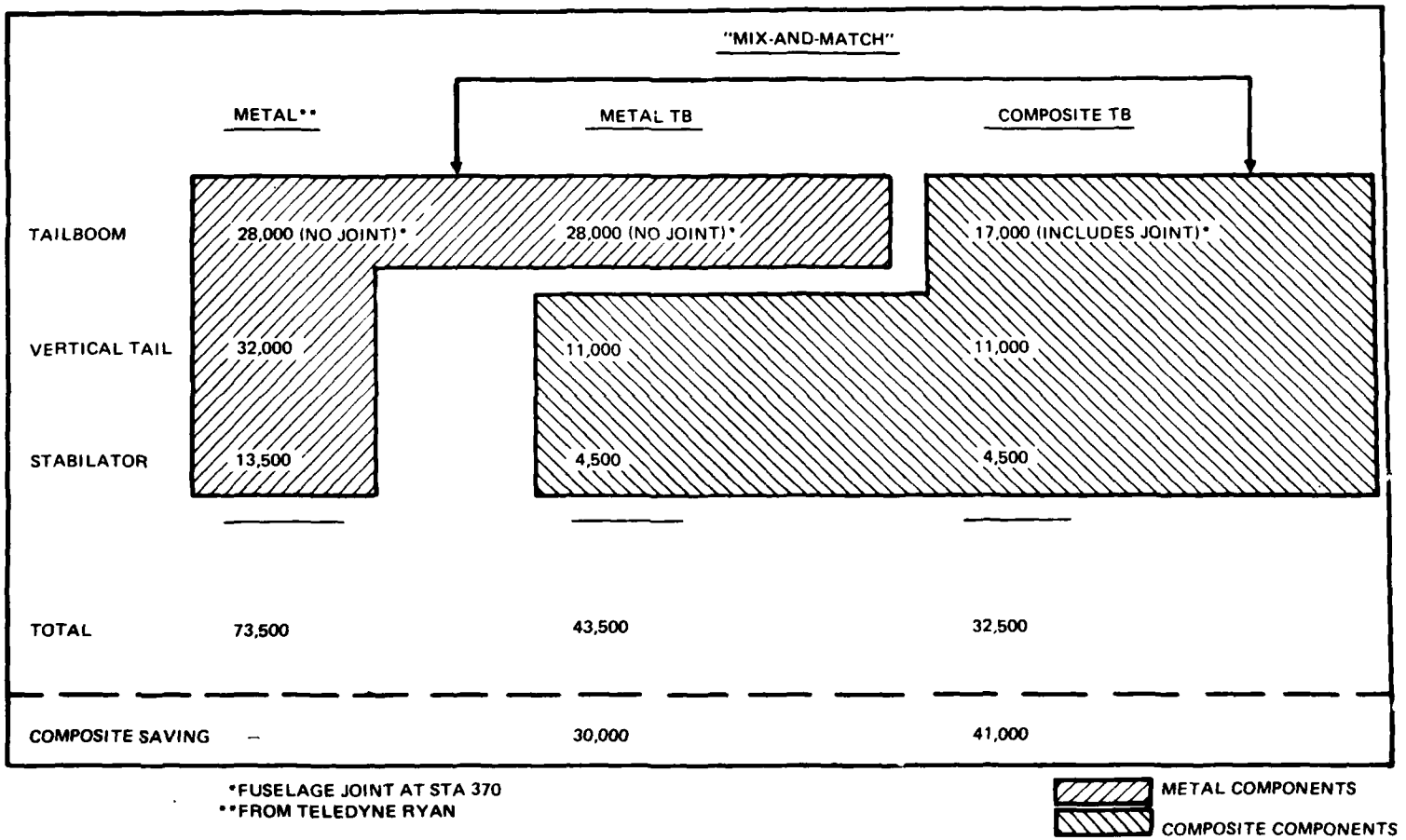


Figure 94. Unit Production Cost Comparison 536 Shipsets - Second Quarter 1981 Dollars

CONCLUSIONS

This MM&T program demonstrated how advanced composites technology can be applied to the tail section of the AH-64A helicopter to realize important performance, weight, and cost benefits. Methods that HHI has established are directly applicable to the CTS with the manufacturing refinement segment of the program having shown the way to build this particular configuration.

The CTS structural design is complete, and the tooling design is nearly so. It is recommended that the fabrication and test activities be reinstated in a timely manner so that the CTS can be qualified for installation on the production AH-64A. For maximum savings, the complete CTS offers:

- 71 pounds weight saving
- $41,000 DTUPC saving (1981 dollars)
- 269 parts count saving
- 9047 fastener count saving
- $1.1 million fuel saving over the life of the fleet
- $22 million overall production and life cycle cost saving for the fleet
- 55 fpm vertical rate of climb increase
- Improved reliability and maintainability

As an alternative to funding the entire CTS, a program limited to only the composite stabilator can offer:

- 19 pounds weight saving
- $9000 DTUPC saving (1981 dollars)
- 11 parts count saving
- 41 fastener count saving
- $300,000 fuel saving over the life of the fleet
- $5.9 million overall production and life cycle cost saving for the fleet
- 15 fpm vertical rate of climb increase
- Improved reliability and maintainability

REFERENCES

1. Needham, J.F., USAAMRDL TR-76-24: Design, Fabrication, and Testing of an Advanced Composite AH-1G Tail Section (Tailboom/ Vertical Fin), February 1976.

2. Head, R.E., USAAMRDL TR-77-19: Flight Test of a Composite Multi-Tubular Spar Main Rotor Blade on the AH-1G Helicopter, August 1977.

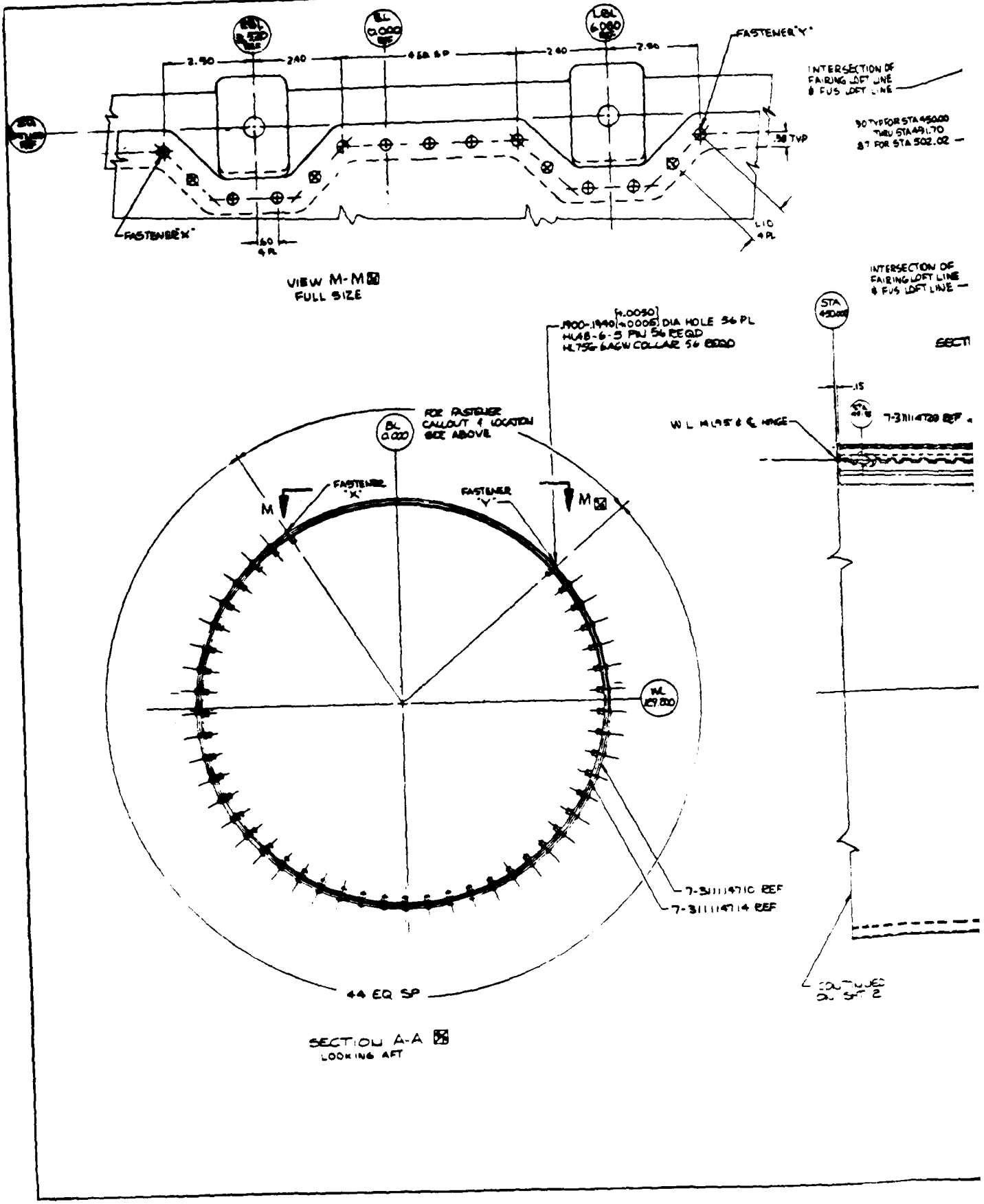
3. Goodall, R.E., USAAMRDL TR-77-27: Advanced Technology Helicopter Landing Gear, October 1977.

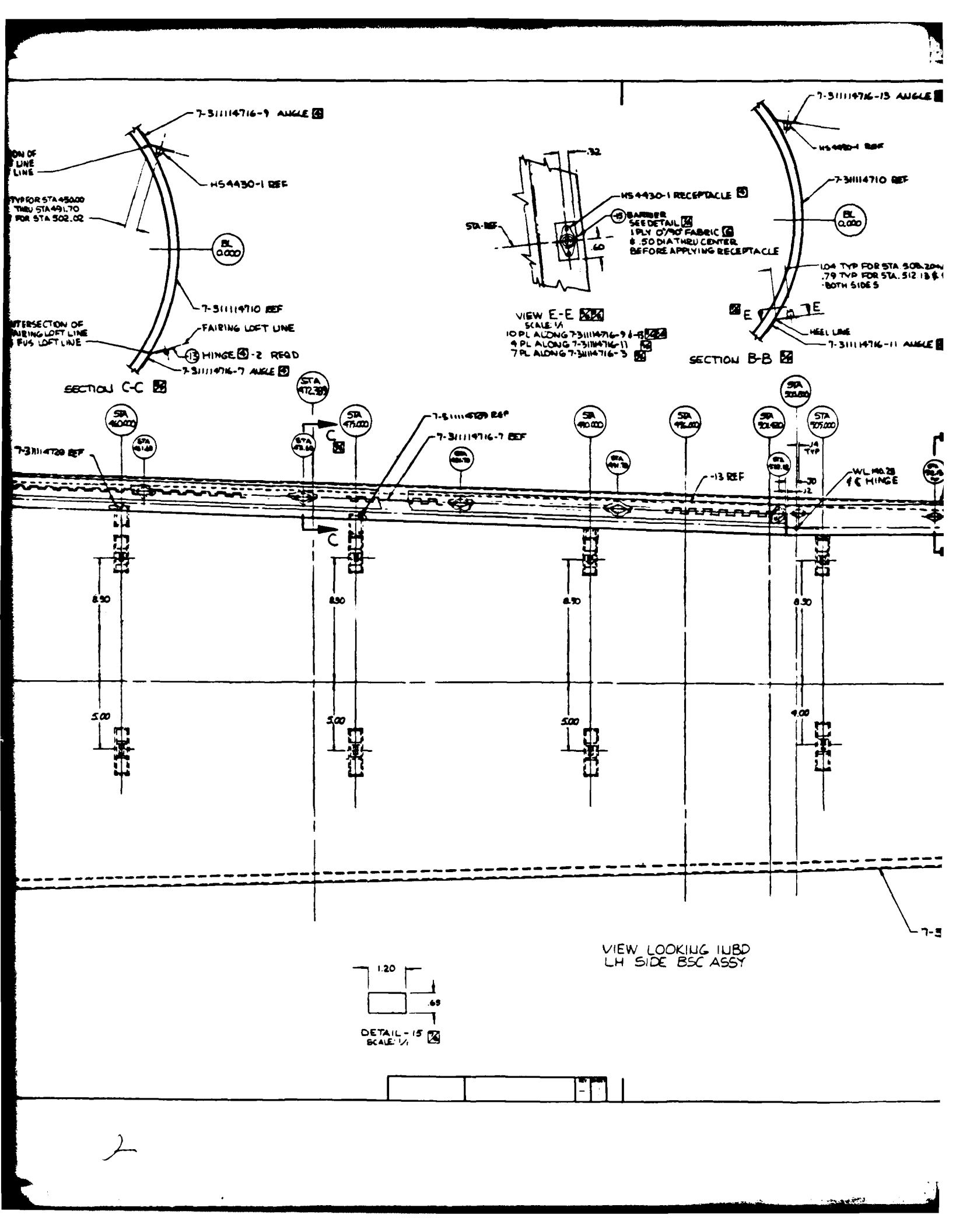
4. Anon, Design Criteria Report for the YAH-64 Advanced Attack Helicopter, Phase 2 Hughes Helicopters, Inc. Report HH 78-174, revised 15 February 1980.

5. MIL-S8698 (ASG) Military Specification Structural Design Requirements, Helicopters, 28 February 1958.

CTS ASSEMBLY AND SUBASSEMBLY DRAWINGS

Figure	Title	Drawing Number
95	Tailboom Assembly	7-311114700, Sheet 1
	Tailboom Assembly	Sheet 2
	Tailboom Assembly	Sheet 3
96	Tailboom Subassembly	7-311114710, Sheet 1
	Tailboom Subassembly	Sheet 2
	Tailboom Subassembly	Sheet 3
	Tailboom Subassembly	Sheet 4
97	Vertical Tail Assembly	7-311122700, Sheet 1
98	Vertical Tail Subassembly	7-311122710, Sheet 1
	Vertical Tail Subassembly	Sheet 2
99	Stabilator Assembly	7-311123710, Sheet 1
	Stabilator Assembly	Sheet 2


FASTENER "Y"
INTERSECTION OF FAIRING LOFT LINE & FUS LOFT LINE
FASTENER "X"
LID 4 PL
VIEW M-M
FULL SIZE
.1900-.1940 [4.0030-4.0085] DIA HOLE 56 PL
HL48-6-5 PIN 56 REQD
HL756 BAGW COLLAR 56 REQD
FOR FASTENER CALLOUT & LOCATION SEE ABOVE
FASTENER "X"
FASTENER "Y"
7-3111147C REF
7-3111147A REF
44 EQ SP
SECTION A-A
LOOKING AFT
W L HL95 & HINGE
7-3111147B REF


7-311114716-9 ANGLE
HS4430-1 REF
TYP FOR STA 450.00 THRU STA 491.70 FOR STA 502.02
BL 0.000
7-311114710 REF
FAIRING LOFT LINE
13 HINGE 4 -2 REQD
7-311114716-7 ANGLE
SECTION C-C
.32
HS4430-1 RECEPTACLE
13 BARRIER SEE DETAIL
1 PLY 0°/90° FABRIC
Ø .50 DIA THRU CENTER BEFORE APPLYING RECEPTACLE
.60
VIEW E-E
SCALE 1/1
10 PL ALONG 7-311114716-9 & -13
9 PL ALONG 7-311114716-11
7 PL ALONG 7-311114716-5
7-311114716-13 ANGLE
7-311114710 REF
BL 0.000
1.04 TYP FOR STA 502.20
.79 TYP FOR STA. 512.13
BOTH SIDES
HEEL LINE
7-311114716-11 ANGLE
SECTION B-B
STA 460.000
STA 472.382
STA 475.000
STA 490.000
STA 495.000
STA 500.000
STA 505.000
7-311114720 REF
7-311114716-7 REF
-13 REF
WL 490.28 ℄ HINGE
.14 TYP
.30
.12
8.50
5.00
4.00
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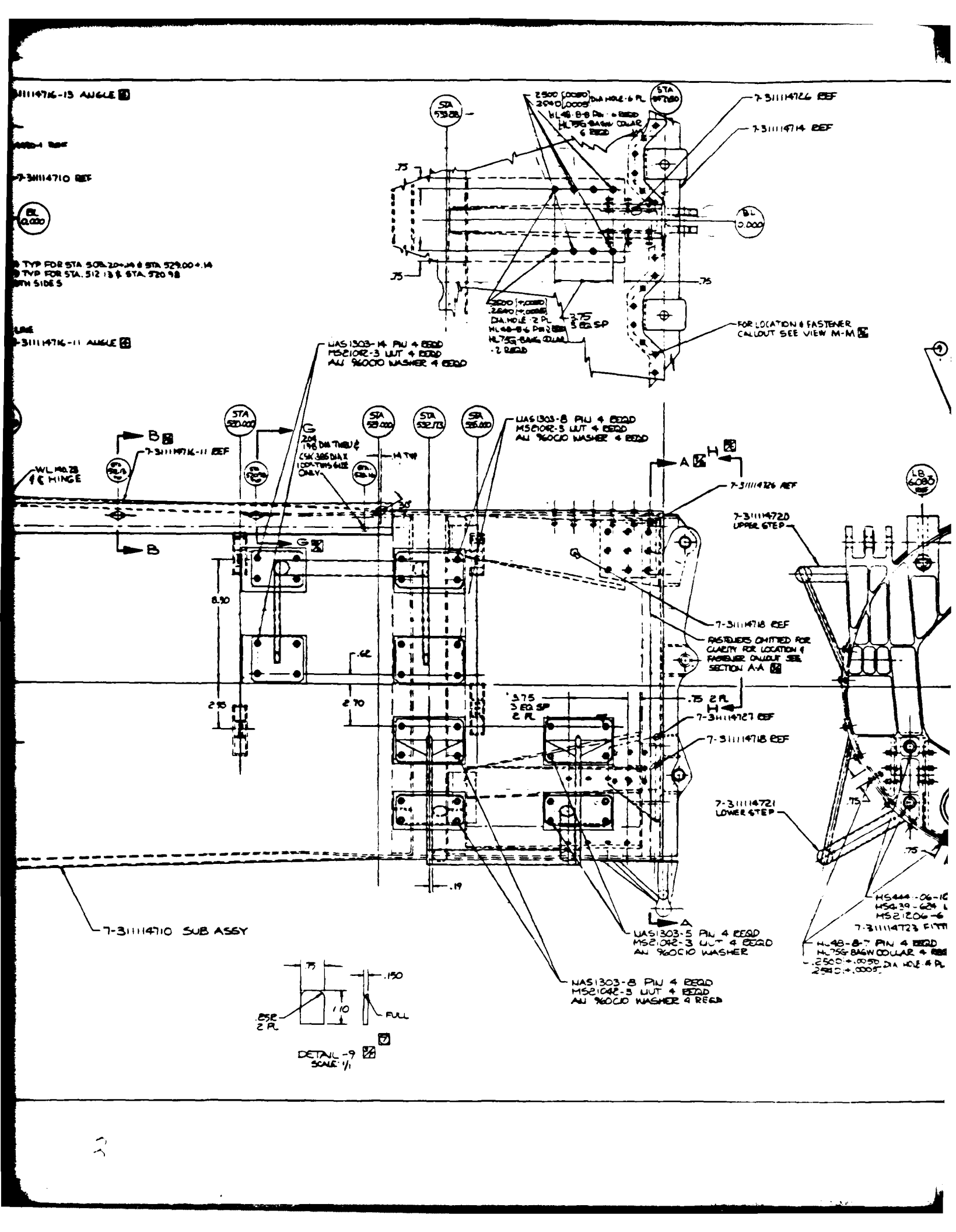
LH SIDE BSC ASSY
1.20
.69
DETAIL - 15
SCALE 1/1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7-311114710 REF
BL 0.000
TYP FOR STA 508.20 & STA 529.00
TYP FOR STA. 512.13 & STA. 520.98
7-311114716-11 ANGLE
STA 539.28
STA 497.80
7-311114726 REF
7-311114714 REF
.75
.375 5 EQ SP
FOR LOCATION & FASTENER CALLOUT SEE VIEW M-M
NAS1303-14 PIN 4 REQD
MS21042-3 NUT 4 REQD
AN 960C10 WASHER 4 REQD
STA 520.000
STA 529.000
STA 532.72
STA 535.000
NAS1303-8 PIN 4 REQD
MS21042-5 NUT 4 REQD
AN 960C10 WASHER 4 REQD
WL 198.28 ℄ HINGE
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7-311114726 REF
7-311114720 UPPER STEP
7-311114718 REF
FASTENERS OMITTED FOR CLARITY FOR LOCATION & FASTENER CALLOUT SEE SECTION A-A
.375 3 EQ SP 2 PL
.75 2 PL
7-311114727 REF
7-311114718 REF
7-311114721 LOWER STEP
7-311114710 SUB ASSY
NAS1303-5 PIN 4 REQD
MS21042-3 NUT 4 REQD
AN 960C10 WASHER
NAS1303-8 PIN 4 REQD
MS21042-5 NUT 4 REQD
AN 960C10 WASHER 4 REQD
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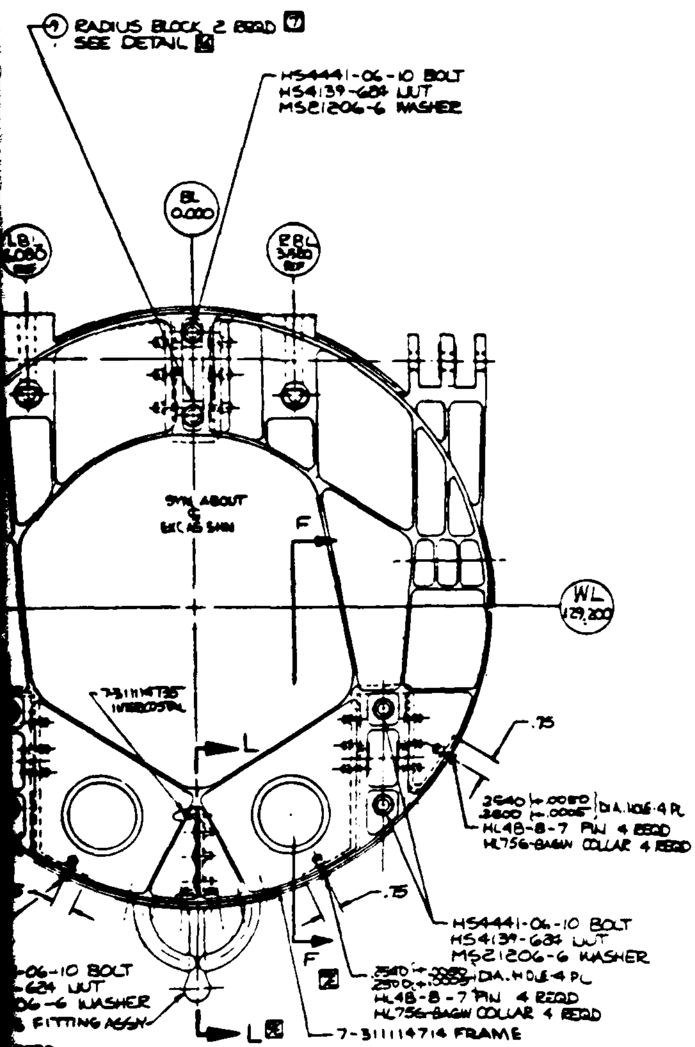
.110
.25R 2 PL
FULL
DETAIL-9
SCALE: 1/1



QTY REQD		PART NO.	NAME	SIZE	DESCRIPTION	SPECIFICATION	ZONE
21		HS4430-1	RECEPTACLE				6F
16		MS4447-1032	NUT				2B
6		MS21206-6	WASHER				
16		MS21042-3	NUT				
2		MS21044N3	NUT				5B
6		HS4139-624	NUT				3F
6		HS4441-06-10	BOLT				3F
34		AN960C10	WASHER				25B
2		NAS1635-3-4	SCREW				
4		NAS1303-14	BOLT				
8		NAS1303-8	BOLT				
4		NAS1303-5	BOLT				
16		NAS 190PBE7	SCREW				
6		HL48-8-8	PIN				
12		HL48-8-7	PIN				
2		HL48-8-6	PIN				
36		HL48-8-5	PIN				
3		HL48-6-7	PIN				
56		HL48-6-5	PIN				
56		HL756-8AGW	COLLAR				
59		HL756-6AGW	COLLAR				
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1		7-311114739	FITTING				29E
1		7-311114735	INTERCOSTAL				3D
1		7-311114734	BRACKET ASSY				22D
1		7-311114733	BRACKET				25D
1		7-311114731	BRACKET				23D
[illegible]		7-311114729	BRACKET				17B
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1		7-311114716-9	FAIRING ANGLE				8H
1		7-311114716-7	FAIRING ANGLE				8F
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1		7-311114710-1	SUB ASSY				5B
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1		-9	BARRIER		5		22D
1		-7	BARRIER		5		22E
21		-15	BARRIER		5		16F
2		-3	HINGE		MS20257-3-3300		5F
1		-11	HINGE		MS20257-3-4000		17F
2		-9	RADIUS BLOCK	20 75 20	2024T3 ALUMINUM	2024-250/4	3F
11		-7	BARRIER		5		[illegible]
44		-3	BARRIER		5		16G

LIST OF MATERIAL

Hughes Helicopters

02731 J

NOTES: UNLESS OTHERWISE SPECIFIED

1 INSTALL HI-LOK FASTENERS PER HP 3-2
2 INSTALL THREADED FASTENERS PER HP6-3
3 DIMENSIONING & TOLERANCING PER ANSI Y14.5
4 FILM ADHESIVE BOND PER HP16-30 WITH HMS 16-1111 FILM ADHESIVE CL 6
5 E-GLASS FABRIC MIL-C-9084, TYPE VIII, GRADE A, CLASS 2. (.010/PLY)
6 HMS 16-1111 PASTE ADHESIVE CL 1.
7 FINISH 12A PER EPB4-230.

Figure 95. Tailboom Assembly (Sheet 1 of 3)

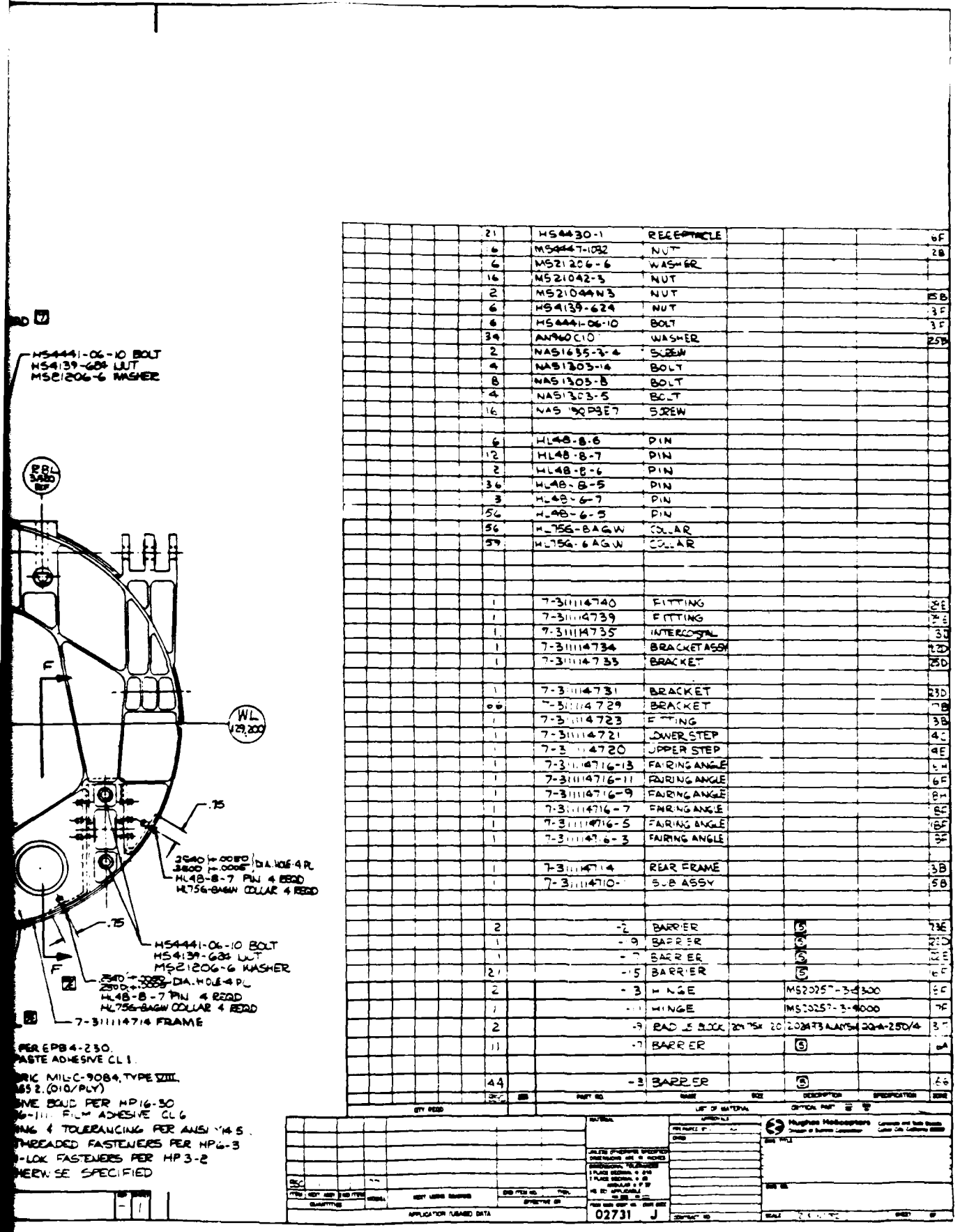
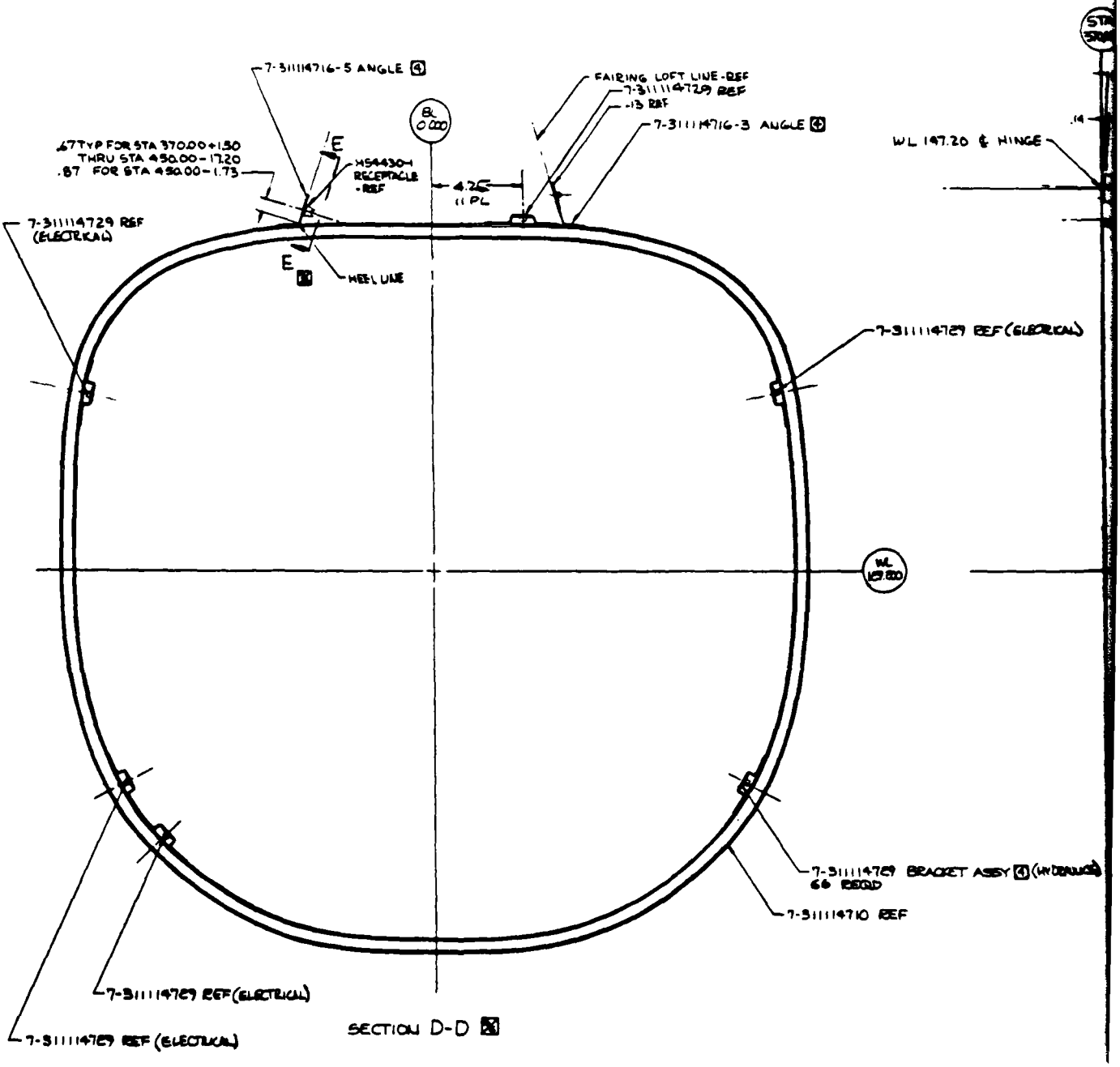


Figure 95. Tailboom Assembly (Sheet 1 of 3)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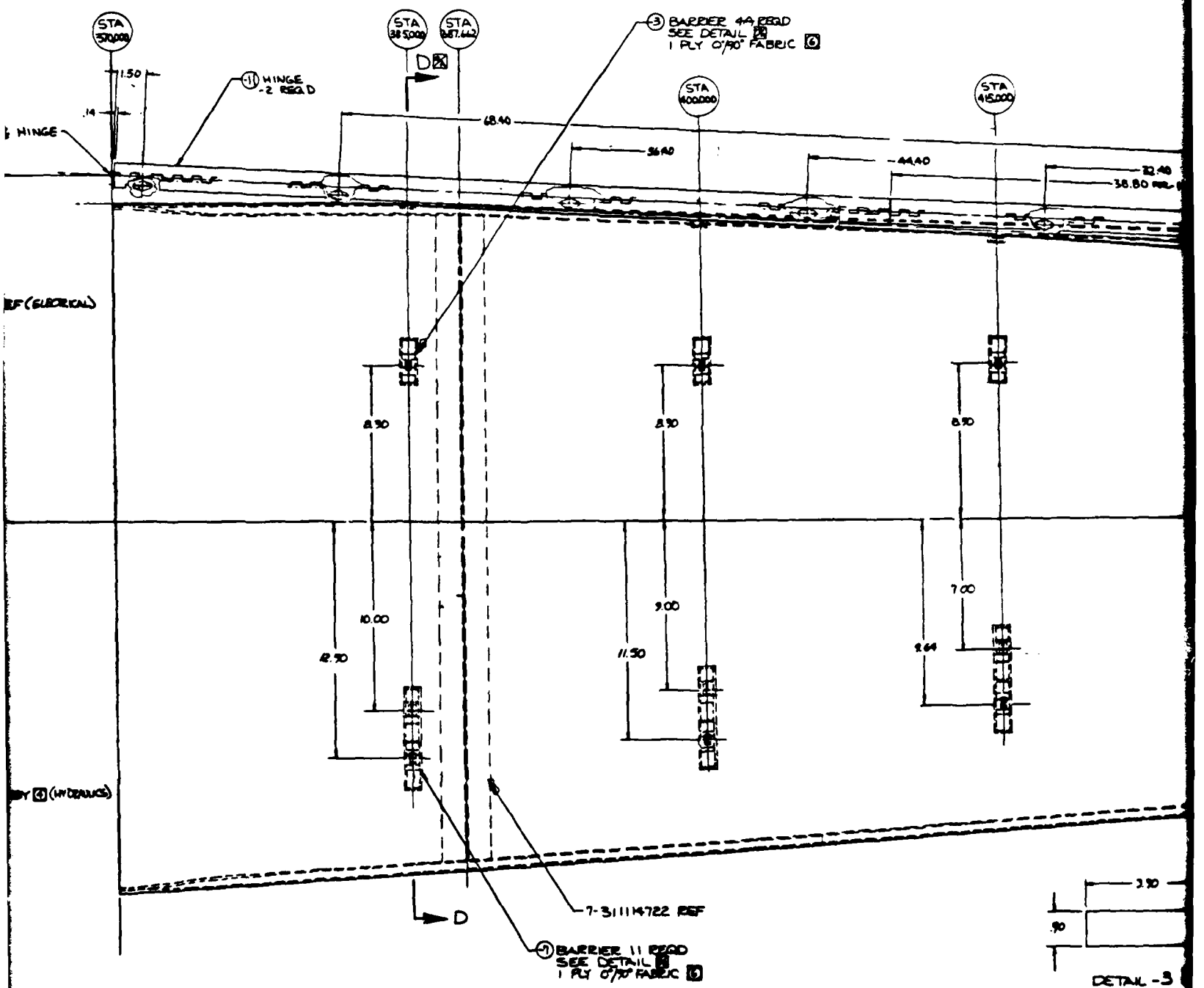
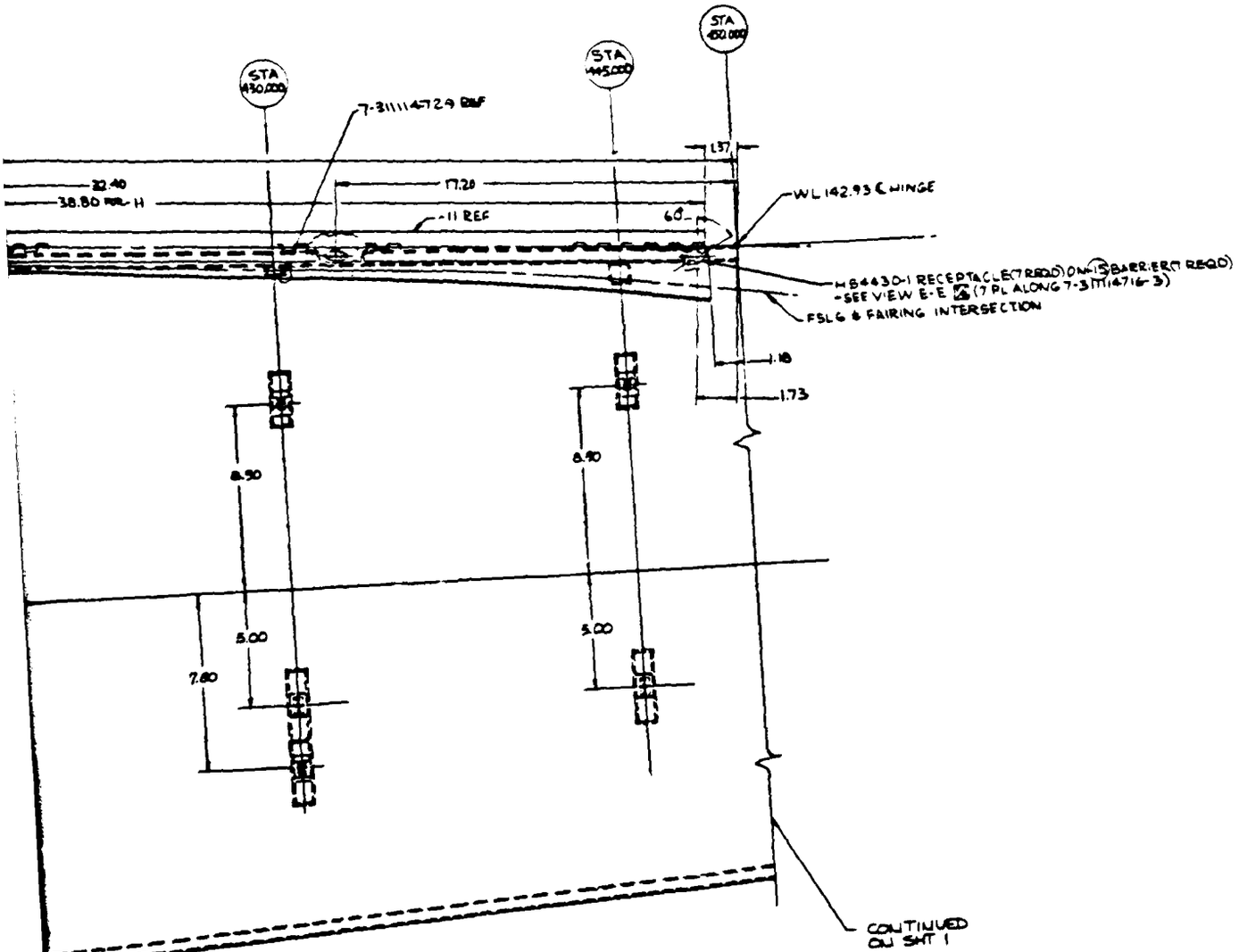
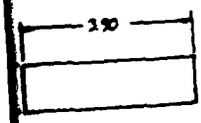
FAIRING LOFT LINE-REF
7-311114729 REF
.13 REF
7-311114716-3 ANGLE 4
WL 147.20 & HINGE
.67 TYP FOR STA 370.00+1.50
THRU STA 450.00-17.20
.87 FOR STA 450.00-1.73
HS44304
RECEPTACLE
-REF
4.25
11 PL
BL 0.000
E
E 10
HEEL LINE
7-311114729 REF
(ELECTRICAL)
7-311114729 REF (ELECTRICAL)
WL 159.00
7-311114769 BRACKET ASSY 4 (W/ DRAWING)
66 REQD
7-311114710 REF
7-311114769 REF (ELECTRICAL)
7-311114769 REF (ELECTRICAL)

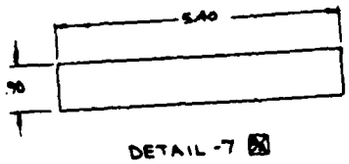
SECTION D-D 8


STA 390.000
STA 387.442
STA 400.000
STA 415.000
(3) BARRIER 44 REQD SEE DETAIL [B] 1 PLY 0/90° FABRIC [6]
(11) HINGE -2 REQD
HINGE
1.50
.14
68.40
56.40
44.40
32.40
38.80
D
D
8.90
8.90
8.90
10.00
9.00
7.00
12.50
11.50
2.64
(7) 3111 4722 RSF
(1) BARRIER 11 REQD SEE DETAIL [B] 1 PLY 0/90° FABRIC [6]
3.90
.90
DETAIL -3





DETAIL -3



DETAIL -7

142.93 C HINGE

MB4630-1 RECEPTACLE (7 REQD) ON (5) BARRIER (7 REQD)
SEE VIEW E-E (7 PL ALONG 7-3 TT14716-3)
& 6 FAIRING INTERSECTION

NTINUED
SHT 1

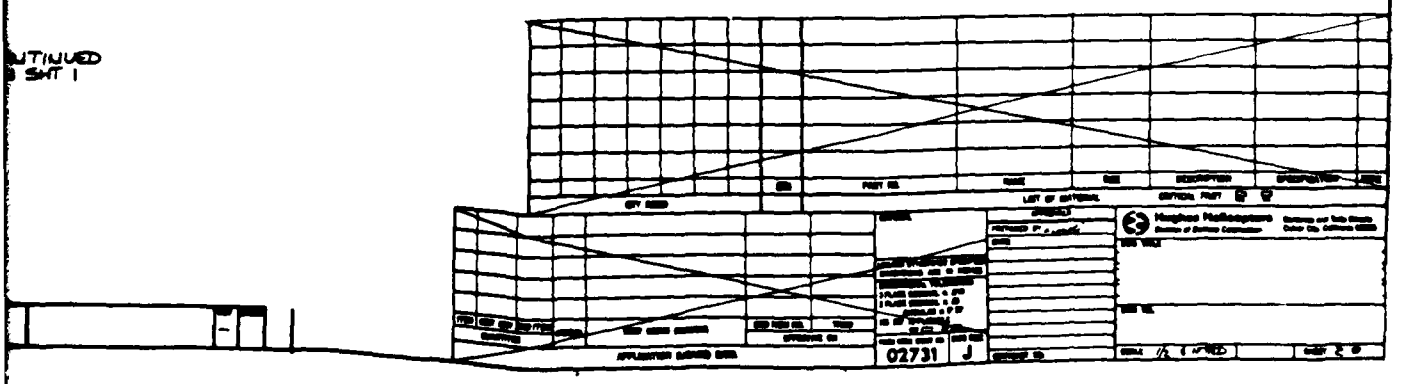
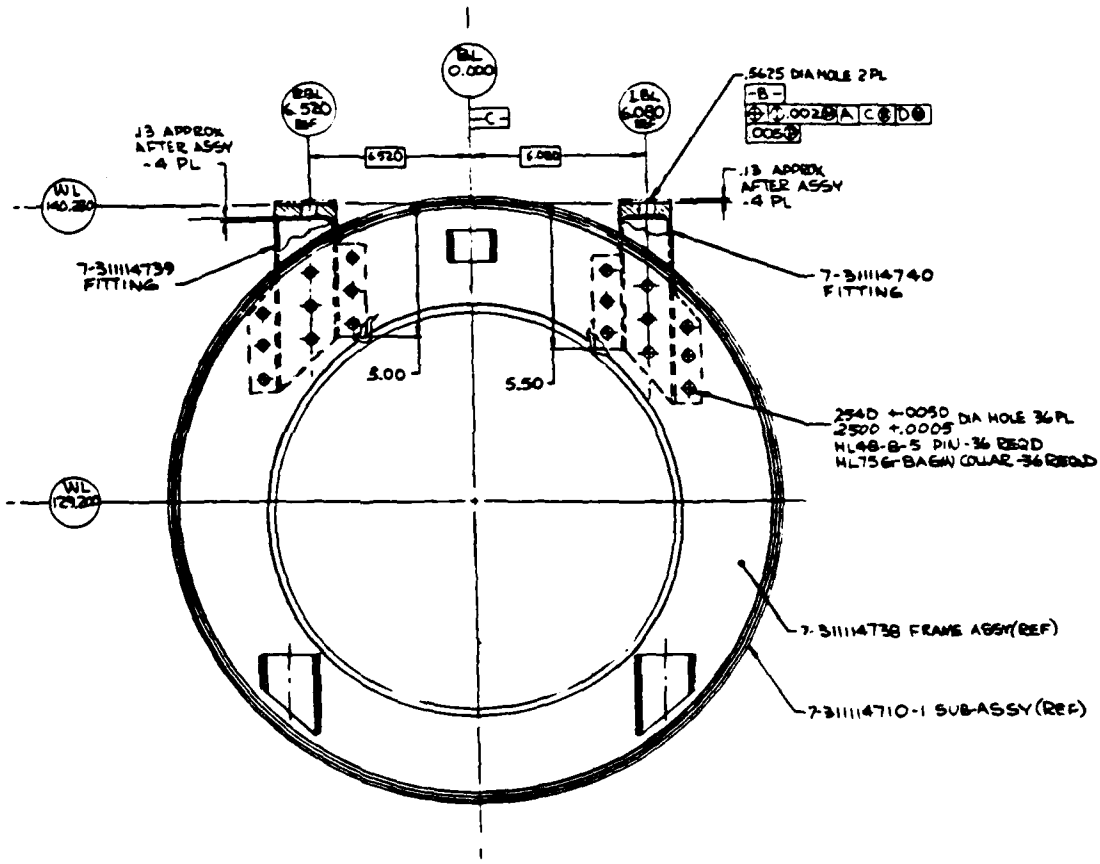
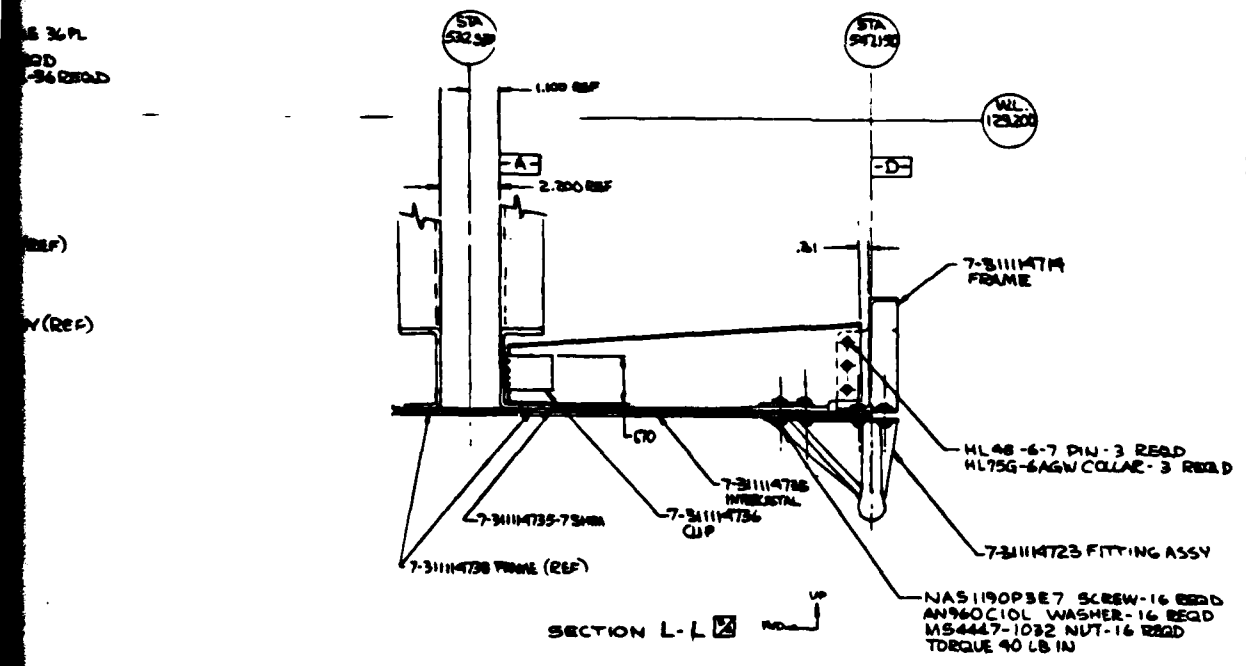
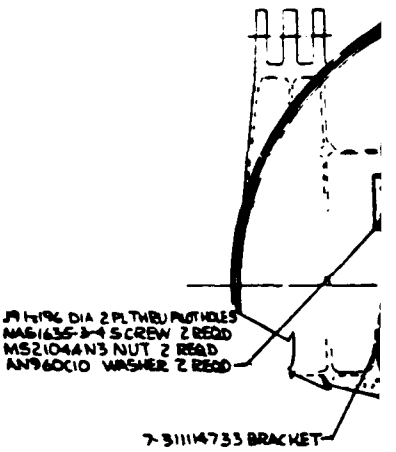


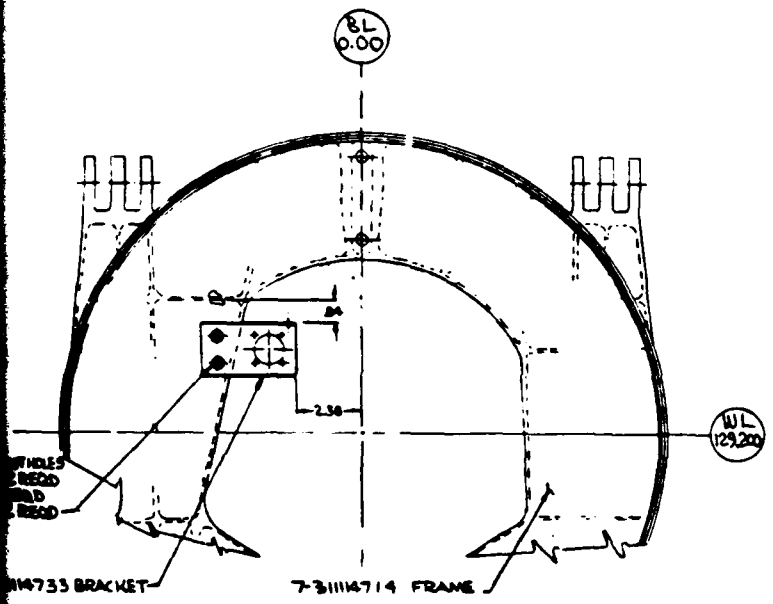
Figure 95. Tailboom Assembly (Sheet 2 of 3)



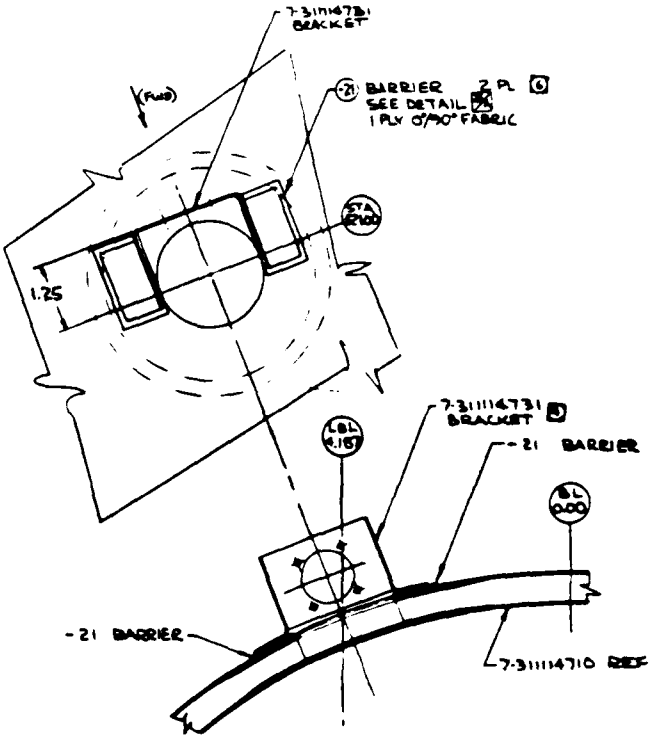
SECTION J-J
LOOKING FWD


STA 532.500
STA 542.150
WL 129.200
1.100 REF
2.200 REF
.81
7-311114719 FRAME
HL48-6-7 PIN-3 REQD
HL75G-6AGW COLLAR-3 REQD
7-311114723 FITTING ASSY
7-311114738 FRAME (REF)
7-311114735-7 SHIM
7-311114736 CLIP
7-311114738 INTERCOSTAL
NAS1190P3E7 SCREW-16 REQD
AN960C10L WASHER-16 REQD
MS4447-1032 NUT-16 REQD
TORQUE 90 LB IN
SECTION L-L
UP
FWD

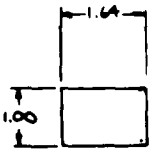

.19 H196 DIA 2 PL THRU PILOT HOLES
MA61635-3-4 SCREW 2 REQD
MS21044N3 NUT 2 REQD
AN960C10 WASHER 2 REQD
7-311114733 BRACKET



SECTION H-H
LOOKING AFT



SECTION G-G
SCALE 1/1
LOOKING AFT



DETAIL -21
SCALE 1/1

DI

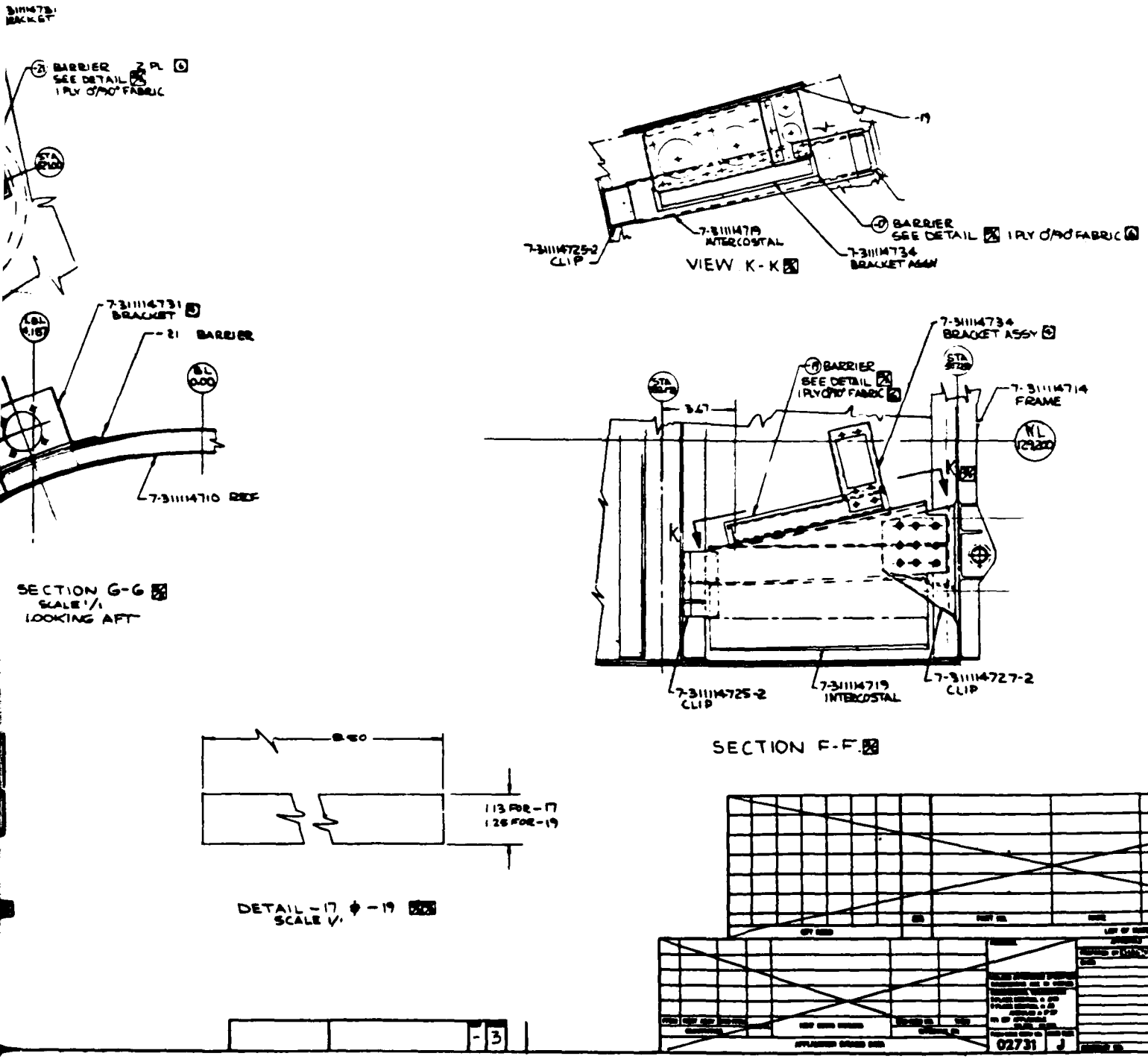
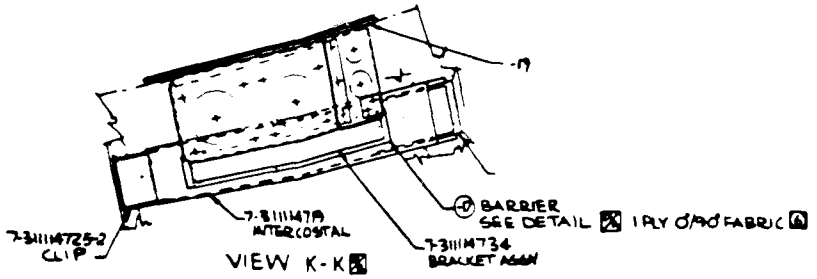
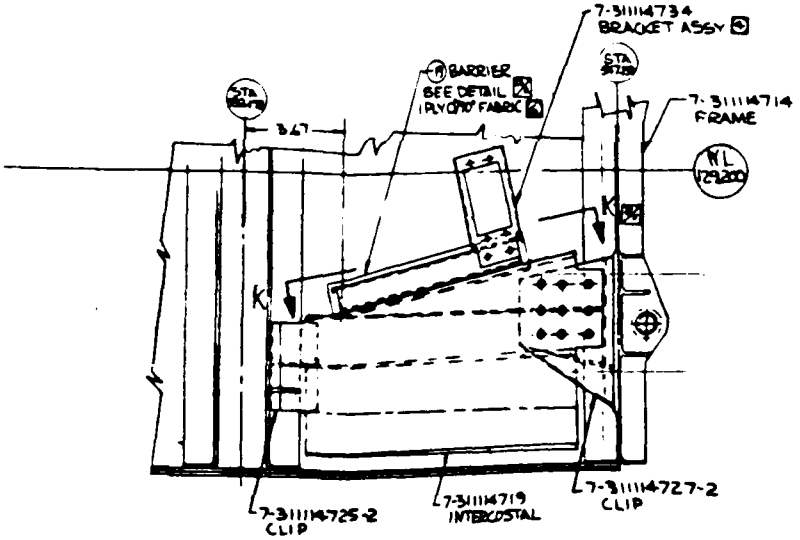


Figure 95. Tailboom Assem





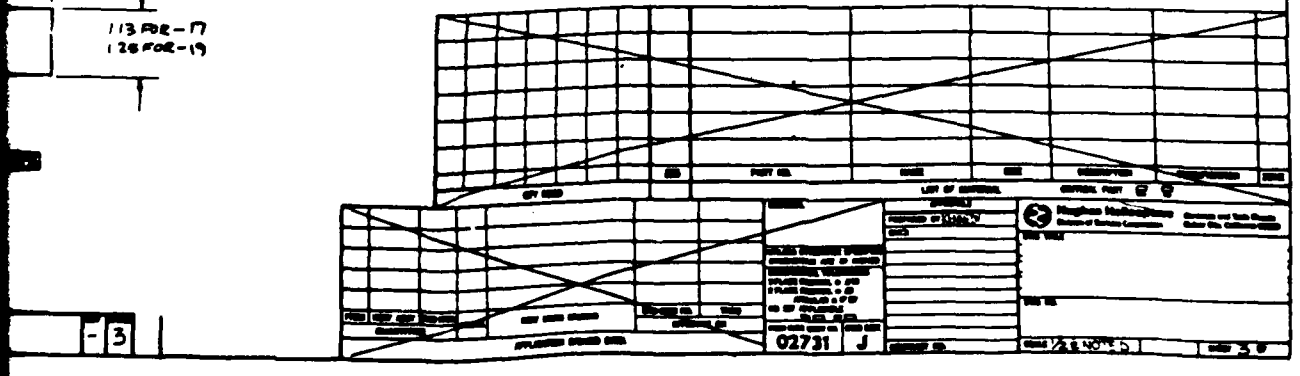
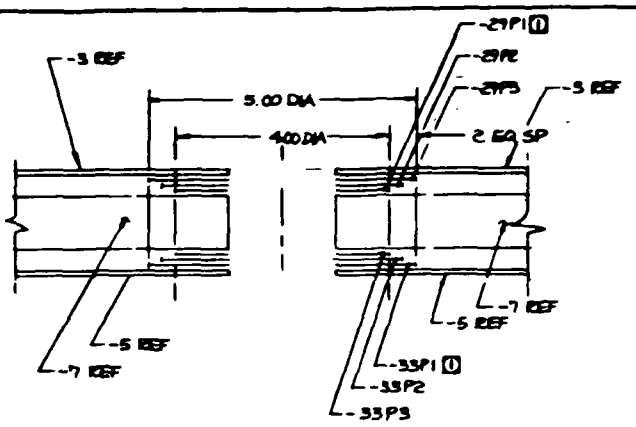
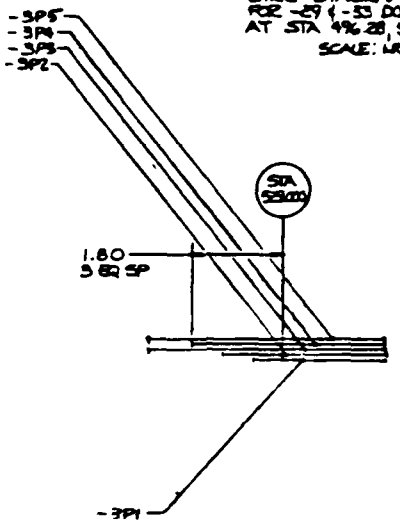


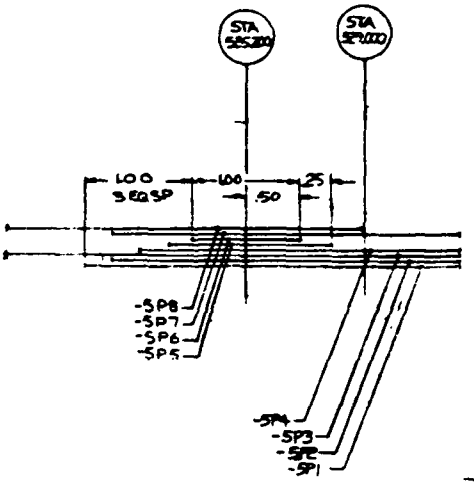
Figure 95. Tailboom Assembly (Sheet 3 of 3)



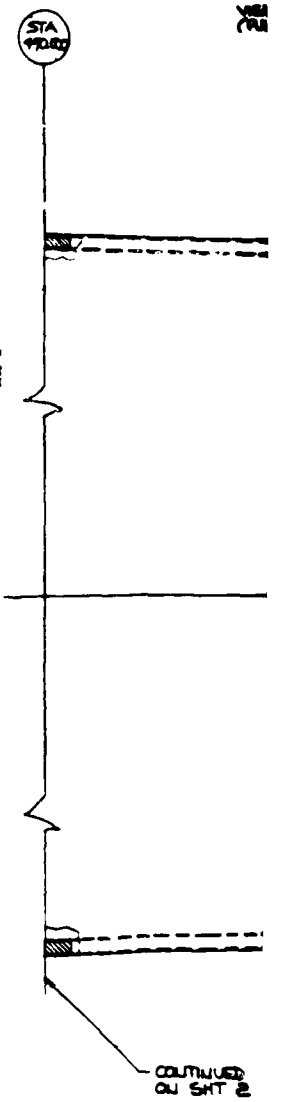
LINE DIAGRAM II
FOR -29 & -33 DOUBLER
AT STA 496.28, 501.70 & 521.00
SCALE: NONE



LINE DIAGRAM I
-3 OUTER SKIN JOINT
@ STA 529.000
NO SCALE



LINE DIAGRAM II
-5 INNER SKIN JOINT
@ STA 529.000
NO SCALE



P3	±45°	6	8	.017
P2	0/90°	6	8	.017
P1	0/90°	6	8	.017
PLY NO	ORIENT	MATL		PLY THK

STACKING SEQUENCE
FOR -29 DOUBLER
AT STA 496.28, 501.70 & 521.00
TABLE 4

P3	0/90°	6	8	.017
P2	0/90°	6	8	.017
P1	±45°	6	8	.017
PLY NO	ORIENT	MATL		PLY THK

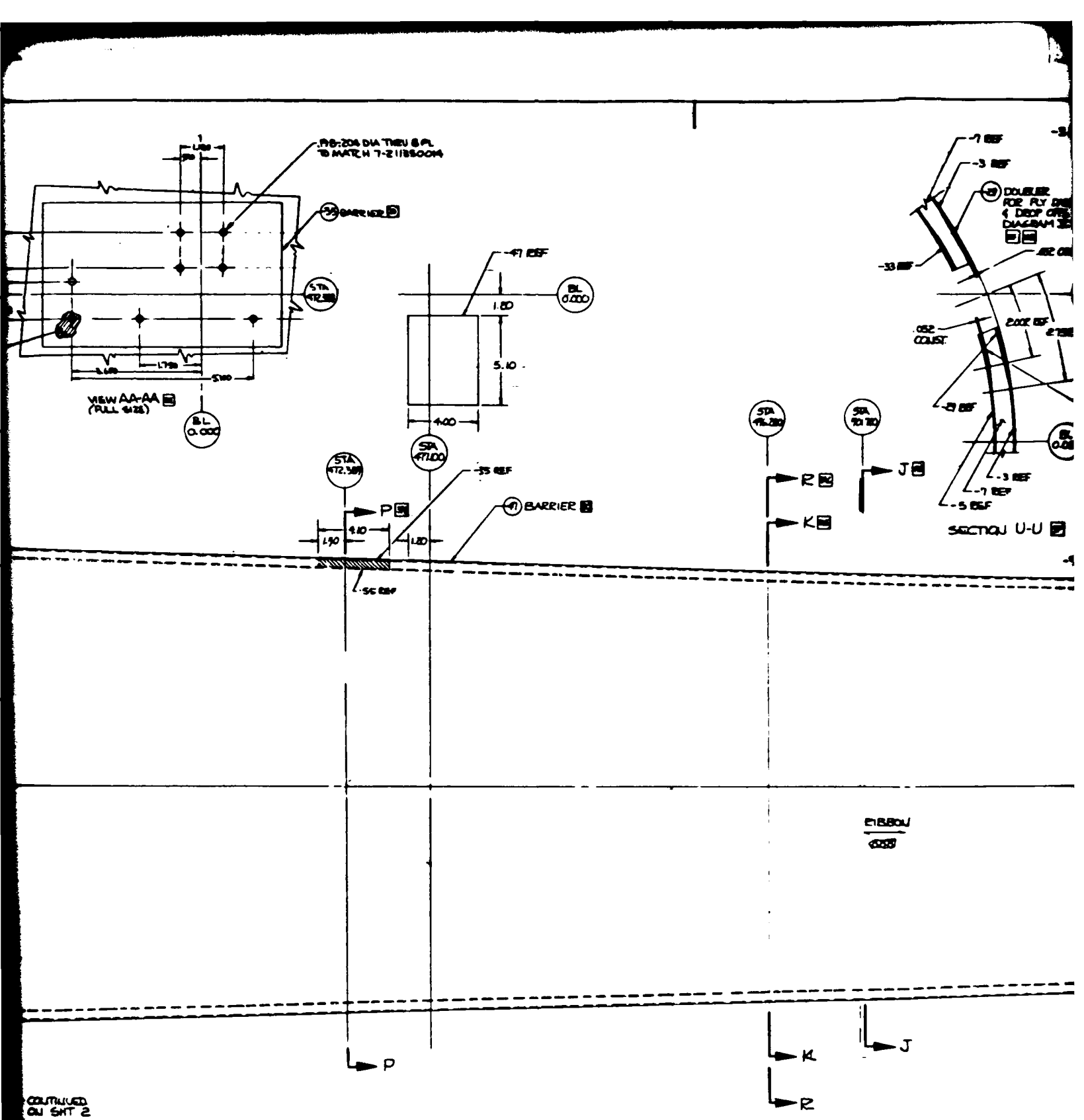
STACKING SEQUENCE
FOR -3 DOUBLER
AT STA 496.28 501.70 & 521.00
TABLE 3

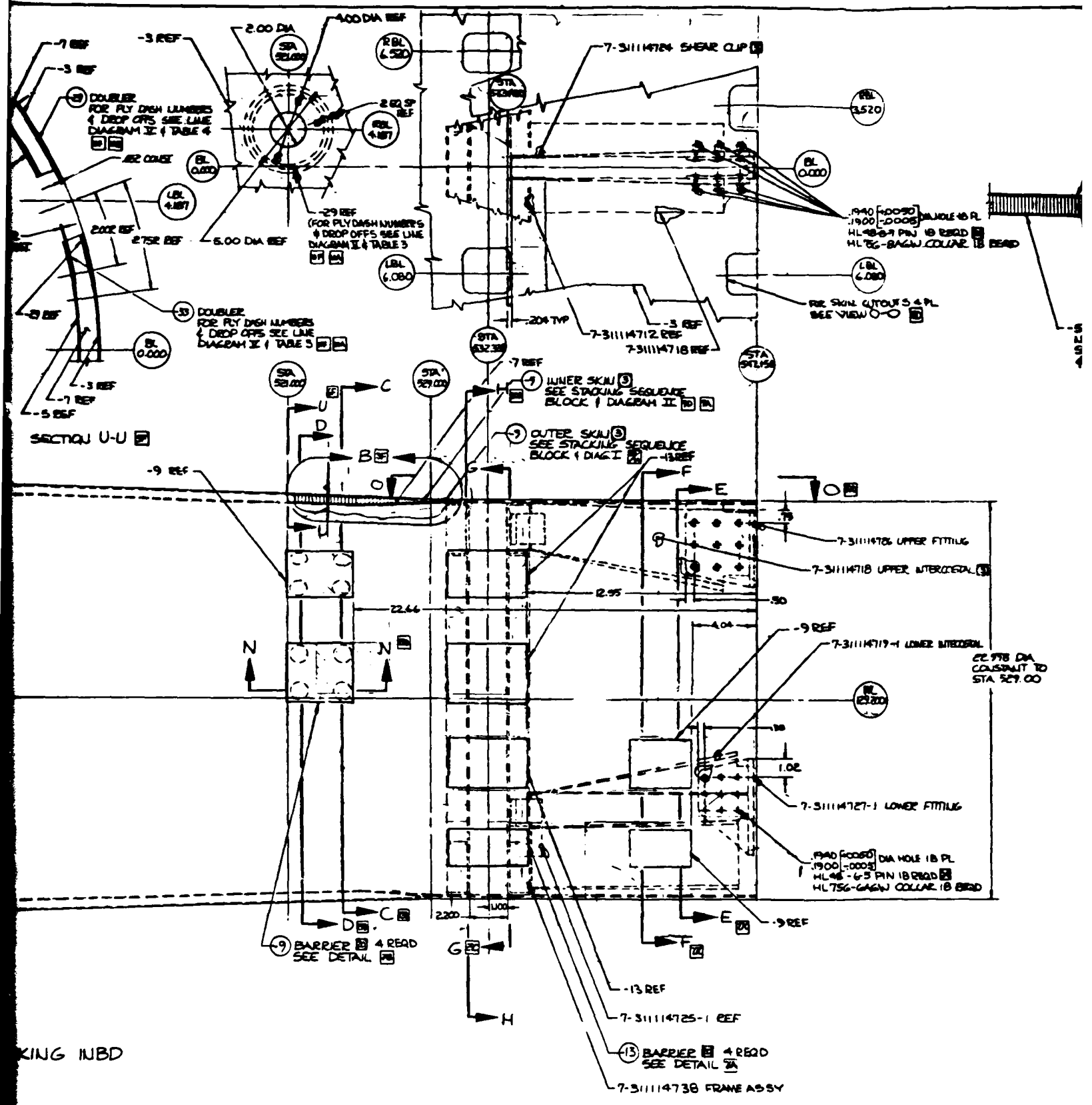
P5	90°	1	3	.010
P4	±45°	5		.017
P3	± 5°	3		.020
P2	±45°	5		.017
P1	±45°	5	3	.017
PLY NO	ORIENT	MATL		PLY THK

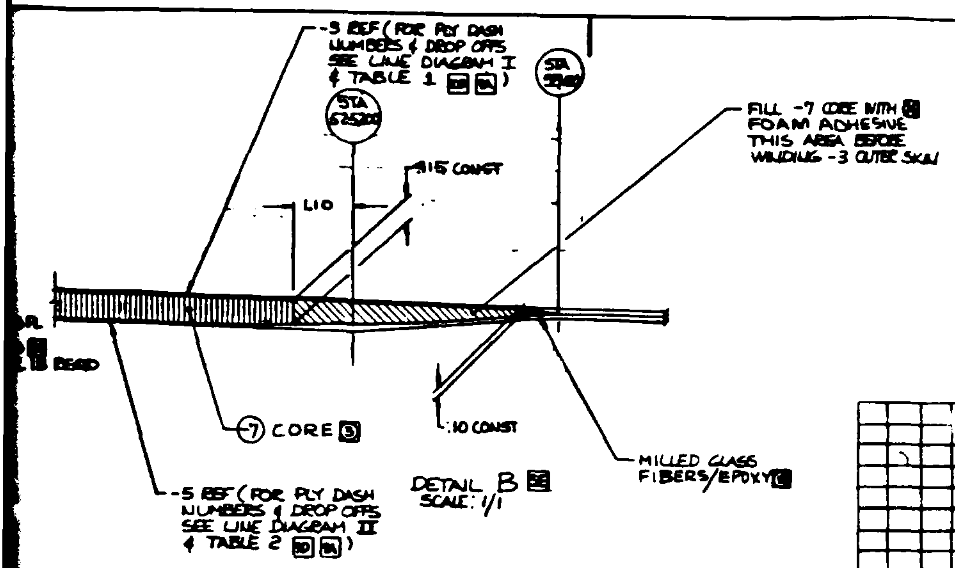
STACKING SEQUENCE
FOR -3 OUTER SKIN
AT STA 370.00
TABLE 1

P10	90°	1	3	.010
P9	±45°	5		.017
P8	0/90°	6		.017
P7	0/90°	6		.017
P6	±45°	6		.017
P5	±45°	6		.017
P4	0/90°	6		.017
P3	±8°	3		.020
P2	±45°	6		.017
P1	90°	1	3	.010
PLY NO	ORIENT	MATL		PLY THK

STACKING SEQUENCE
FOR -5 INNER SKIN
AT STA 370.00
TABLE 2


.118-.204 DIA THRU & PL
TO MATCH 7-2118500M
BARRIER
STA 472.500
BL 0.000
VIEW AA-AA
(FULL SIZE)
1.80
5.10
4.00
STA 471.000
-47 REF
-35 REF
BARRIER
P
4.10
1.90
1.20
-56 REF
STA 471.280
STA 470.780
R
K
J
SECTION U-U
-7 REF
-3 REF
-33 REF
.052
COAST.
2.000 REF
-29 REF
-3 REF
-7 REF
-5 REF
RIBBON
P
K
J
R
CONTINUED
ON SHT 2
VIEW LOOKING INBD
BSC ASSY


SECTION U-U
7-311114784 SHEAR CLIP
DOUBLER FOR PLY DASH NUMBERS & DROP OFFS SEE LINE DIAGRAM II & TABLE 4
DOUBLER FOR PLY DASH NUMBERS & DROP OFFS SEE LINE DIAGRAM II & TABLE 3
INNER SKIN SEE STACKING SEQUENCE BLOCK I DIAGRAM II
OUTER SKIN SEE STACKING SEQUENCE BLOCK I DIAGRAM II
7-311114712 REF
7-311114718 REF
7-311114726 UPPER FITTING
7-311114718 UPPER INTERCOSTAL
7-311114719-1 LOWER INTERCOSTAL
7-311114727-1 LOWER FITTING
BARRIER 4 REQD SEE DETAIL
7-311114725-1 REF
7-311114738 FRAME ASSY
FR SKIN CUTOUT 5 4 PL SEE VIEW O-O
CONSTRUCT TO STA 529.00
KING INBD



- [24] SEAL FASTENERS PER HP 5-10 TYPE IV
- [23] HYGRID - 55 LB. DENSITY ALUMINUM MATERIAL - AMERICAN CANNITYUD CO., BLOOMINGDALE PLANT, HAVRE DE GRACE, MD. 21078
- [22] 50% VOLUME GRAPHITE [5] ROVING & 50% VOLUME KEVLAR [4] ROVING (.020 TOTAL PLY THK)
- [21] 50% VOLUME GRAPHITE [5] ROVING & 50% VOLUME KEVLAR [4] ROVING (.010/PLY)
- [20] HMS 16-1068 PASTE ADHESIVE CL 3. APPLY PER HP16-5
- [19] HMS 16-1111 FILM ADHESIVE, CL 3.
- [18] MILLED GLASS FIBERS/EPOXY 1/32 LENGTH THALCO, LA
- [17] FINISH EXTERIOR ONLY CODE E PER EPB 4-230
- 16. IDENTIFY PER HP8-5 TYPE I, CL 3 LOC OPT
- 15. DIMENSIONING & TOLERANCING PER ANSI Y14.5
- [14] FOAM ADHESIVE PER HMS 16-1111, CL 4
- [13] E-GLASS FABRIC MIL-C-9084 TYPE VIII, A CL 2 (.010/PLY)
- 12. LAYUP PER HP 15-42
- [11] CORE (NOMEX) HMS 16-1114 TYPE V, CL 1
- 10 INSTALL HI-LOK FASTENERS PER HP 3-2
- 9 FOR LOFT LINES SEE DWG 7-1L1113101 TO STA 528.00
- [8] EPOXY RESIN SYSTEM HMS 16-1115 TYPE I, CL 1
- [7] E-GLASS ROVING MIL-R-60346 TYPE I. CL 1 (.010/PLY)
- [6] GRAPHITE FABRIC HMS 16-1163 TYPE II, CL 1 GR A (.017/PLY)
- [5] GRAPHITE ROVING HMS 16-1163 TYPE I, CL 1
- [4] KEVLAR ROVING HMS 16-1164 TYPE I, CL 3
- [3] FILM ADHESIVE BOND PER HP 16-30
- 2. FILAMENT WIND PER HP 15-67
- [1] FIRST PLY GOES AGAINST TOOL FACE

NOTES UNLESS OTHERWISE SPECIFIED

						1		-63	DOUBLER	AR	[6][8]
						1		-61	DOUBLER	AR	[6][8]
						1		-59	DOUBLER	AR	[6][7][8]
						1		-57	DOUBLER	AR	[6][7][8]
						18	97215	HL75G-8AGW	COLLAR		
						18	97215	HL75G-6AGW	COLLAR		
						18	97215	HL48-8-7	PIN		
						18	97215	HL48-6-5	PIN		
						AR		HMS 16-1111, CL 3	FILM ADHESIVE		
						4		SL601-3-3CM	INSERT		
						AR		HMS 1068, CL 3	PASTE ADHESIVE		[20]
						2		7-311114737	INSERT		
						1		7-311114727-2	LOWER FITTING		
						1		7-311114727-1	LOWER FITTING		
						1		7-311114726	UPPER FITTING		
						1		7-311114725-2	LOWER CLIP		
						1		7-311114725-1	LOWER CLIP		
						1		7-311114724-2	UPPER CLIP		
						1		7-311114722	FRAME		
						1		7-311114719-2	LOWER INTERCOSTAL		
						1		7-311114719-1	LOWER INTERCOSTAL		
						1		7-311114718	UPPER INTERCOSTAL		
						2		7-311114736	SHEAR CLIP		
						1		7-311114735	INTERCOSTAL		
						1		7-311114738	FRAME ASSY		
						1		-55	INSERT	[illegible]	[23]
						1		-53	DOUBLER		[18]
						2		-51	DOUBLER		[6][8]
						2		-49	DOUBLER		[6][8]
						1		-47	BARRIER		[13]
						1		-45	BARRIER		[13]
						1		-43	BARRIER		[13]
						1		-41	DOUBLER		[13]
						1		-39	BARRIER		[13]
						1		-37	BARRIER		[13]
						1		-35	BARRIER		[13]
						3		-33	DOUBLER		[6][8]
						1		-31	DOUBLER		[13]
						3		-29	DOUBLER		[6][8]
				2	—	—		-27	BARRIER	3/4" Φ	[13]
				1	—	—		-25	PAD	3/4" Φ	[22]
						1		-23	BARRIER		[13]
						21		-21	INSERT ASSY		
					2	—		-19	BARRIER	1" Φ	[13]
					1	—		-17	PAD	1" Φ	[22]
						8		-15	INSERT ASSY		
						4		-13	BARRIER		
						4		-9	BARRIER		[13]
						1		-7	CORE	AR	[11]
						1		-5	INNER SKIN	AR	[21][22][1][3]
						1		-3	OUTER SKIN	AR	[21][22][1][3]
				-21	-15	BSC					

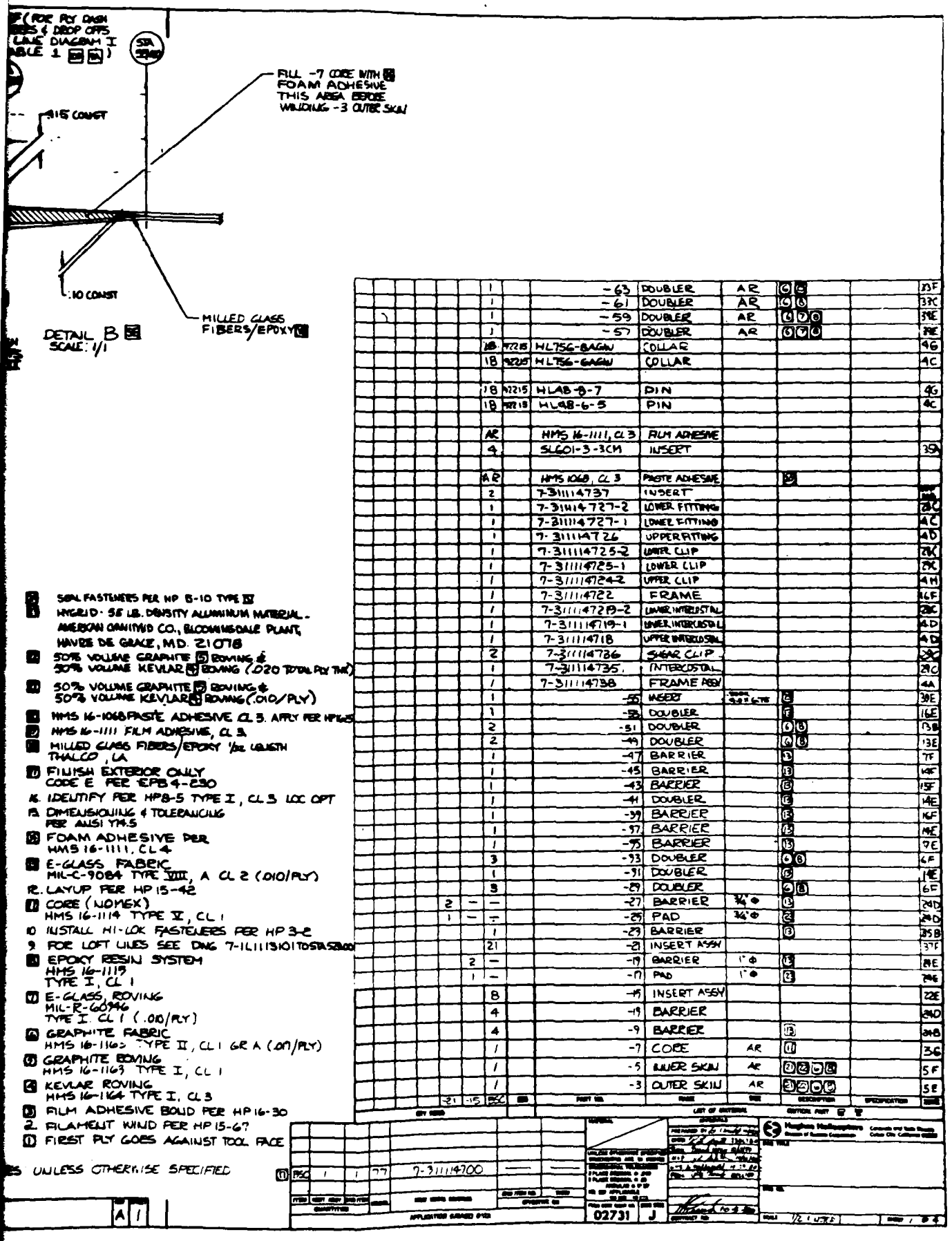
[11] BSC | 1 | 1 | 77 | 7-311114700 | — | —

02731 | J

A | 1

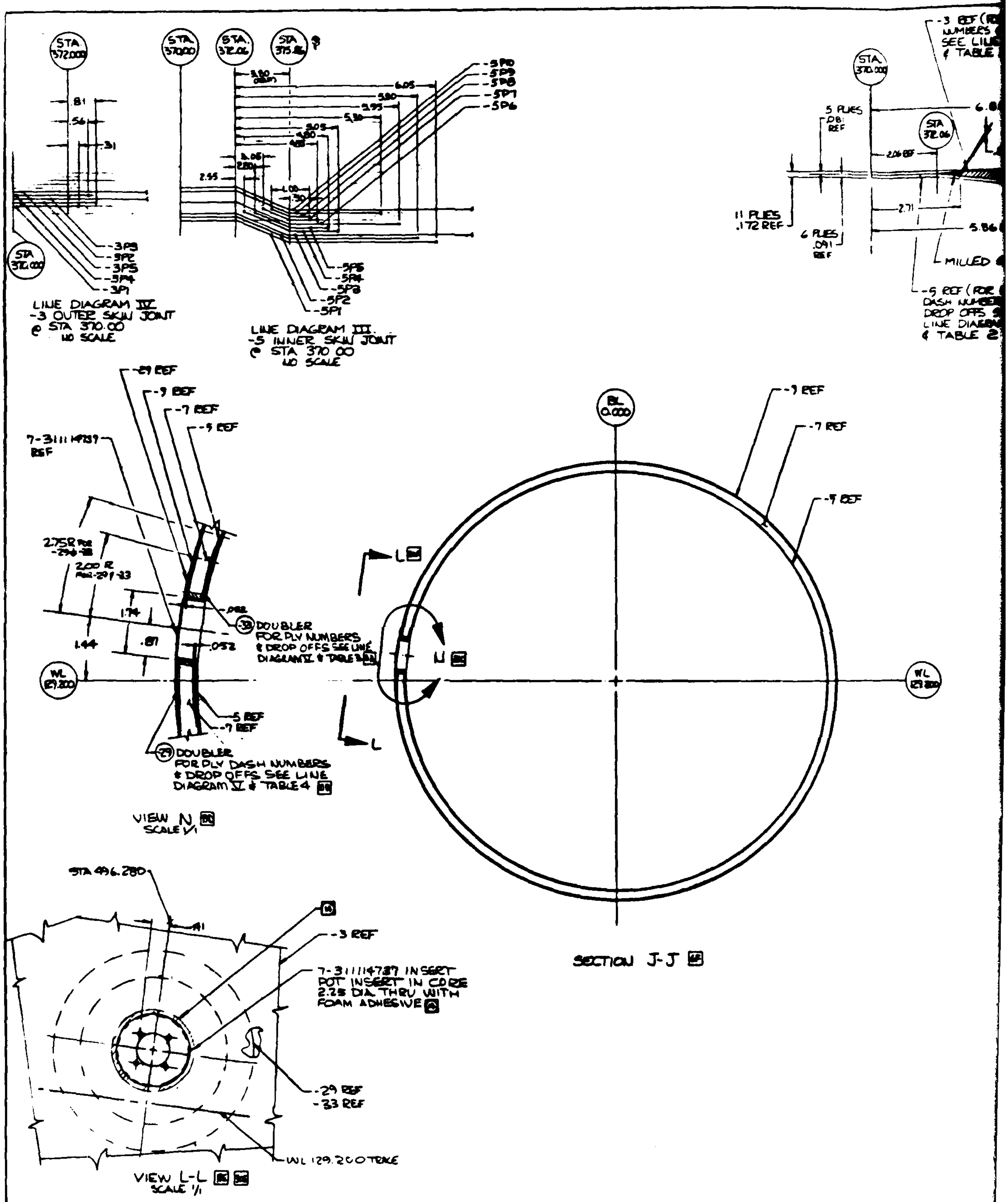
Figure 96. Tailboom Subassembly (Sheet

4

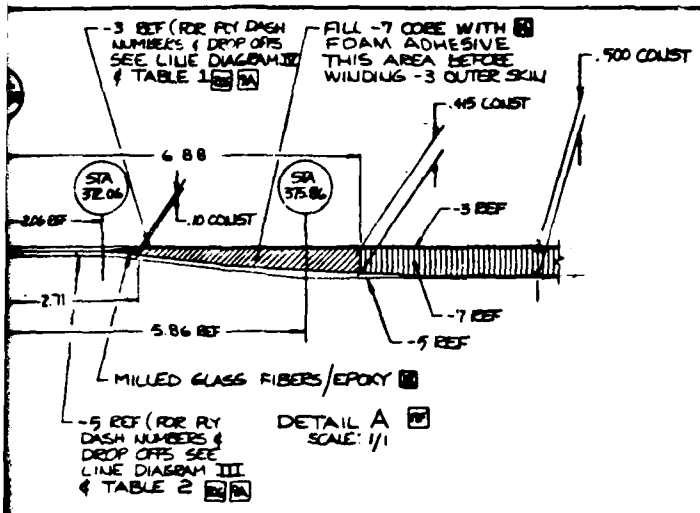


						1			-63	DOUBLER	AR	6 8			33F
						1			-61	DOUBLER	AR	6 8			33C
						1			-59	DOUBLER	AR	6 7 8			33E
						1			-57	DOUBLER	AR	6 7 8			33E
						18	M7215	HL756-6AGW		COLLAR					4G
						18	M7215	HL756-6AGW		COLLAR					4C
						18	M7215	HLAB-8-7		PIN					4G
						18	M7215	HL98-6-5		PIN					4C
						AR		HMS 16-1111, CL 3		FILM ADHESIVE					
						4		SL601-3-3CM		INSERT					35
						AR		HMS 1068, CL 3		PASTE ADHESIVE		5			
						2		7-311114737		INSERT					
						1		7-311114727-2		LOWER FITTING					21C
						1		7-311114727-1		LOWER FITTING					4C
						1		7-311114726		UPPER FITTING					4D
						1		7-311114725-2		LOWER CLIP					21K
						1		7-311114725-1		LOWER CLIP					21K
						1		7-311114724-2		UPPER CLIP					4H
						1		7-311114722		FRAME					16F
						1		7-311114721-2		LOWER INTERCOSTAL					21K
						1		7-311114719-1		LOWER INTERCOSTAL					4D
						1		7-311114718		UPPER INTERCOSTAL					4D
						2		7-311114736		SHEAR CLIP					21K
						1		7-311114735		INTERCOSTAL					21C
						1		7-311114738		FRAME ASSY					4A
						1			-55	INSERT		5			39E
						1			-53	DOUBLER		5			16E
						2			-51	DOUBLER		6 8			13B
						2			-49	DOUBLER		6 8			13E
						1			-47	BARRIER		13			7F
						1			-45	BARRIER		13			14F
						1			-43	BARRIER		13			15F
						1			-41	DOUBLER		6 8			14E
						1			-39	BARRIER		13			14F
						1			-37	BARRIER		13			14E
						1			-35	BARRIER		13			7E
						3			-33	DOUBLER		6 8			6F
						1			-31	DOUBLER		6 8			14E
						3			-29	DOUBLER		6 8			6F
		2	—	—					-27	BARRIER	3/4" Φ	13			21D
		1	—	—					-25	PAD	3/4" Φ	2			21D
						1			-23	BARRIER		13			35B
						21			-21	INSERT ASSY					37F
			2	—					-19	BARRIER	1" Φ	13			14E
			1	—					-17	PAD	1" Φ	2			14G
						8			-15	INSERT ASSY					22E
						4			-13	BARRIER					21D
						4			-9	BARRIER		13			34B
						1			-7	CORE	AR	11			36
						1			-5	INNER SKIN	AR	21 8			5F
						1			-3	OUTER SKIN	AR	21 8			5E
		-21	-15	PSC											

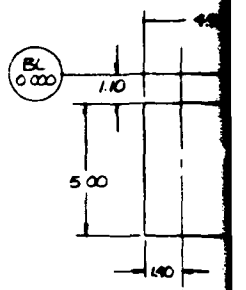
Figure 96. Tailboom Subassembly (Sheet 1 of 4)

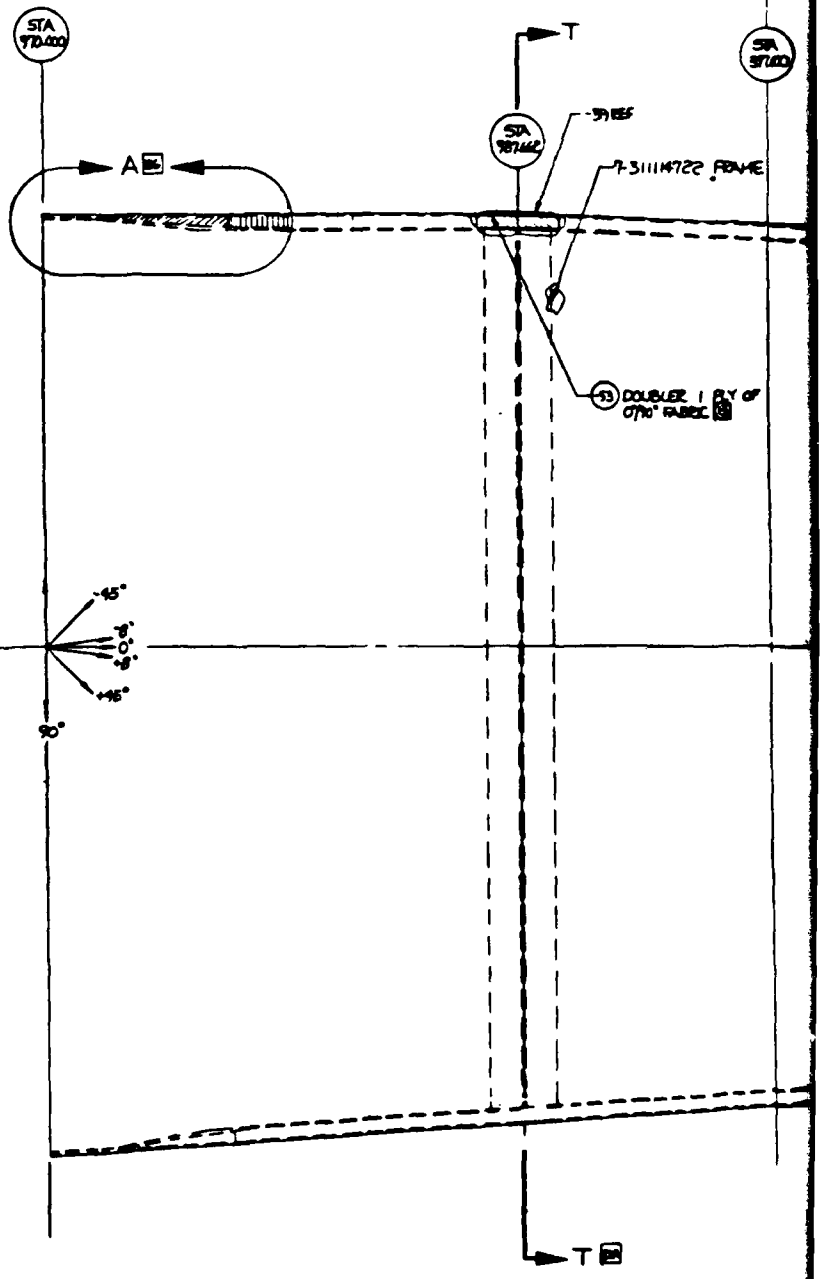


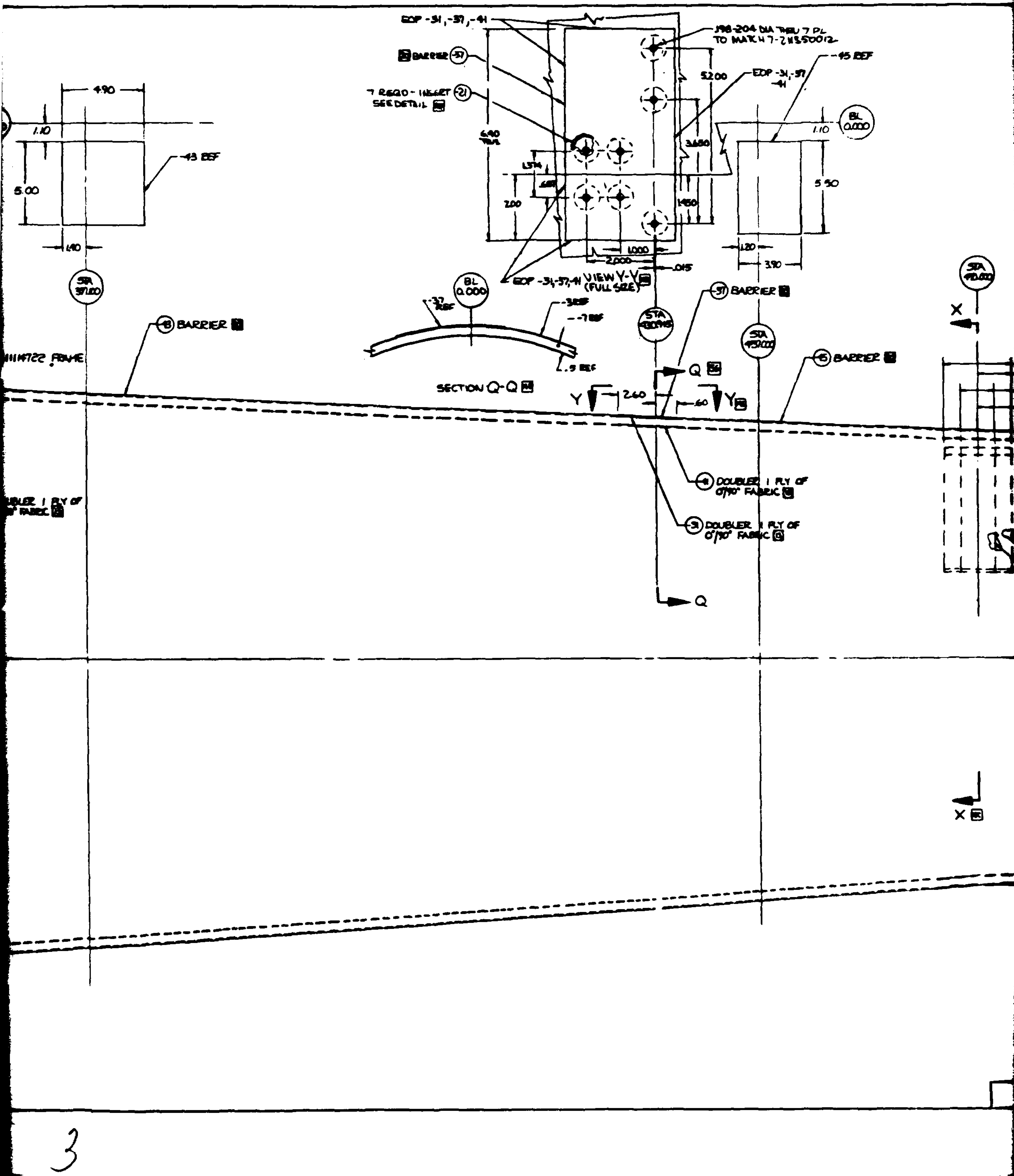
STA 372.000
STA 370.00
STA 372.06
STA 375.86
-5P10
-5P9
-5P8
-5P7
-5P6
-3P3
-3P2
-3P5
-3P4
-3P1
-5P5
-5P4
-5P3
-5P2
-5P1
LINE DIAGRAM IV
-3 OUTER SKIN JOINT
@ STA 370.00
NO SCALE
LINE DIAGRAM III
-5 INNER SKIN JOINT
@ STA 370.00
NO SCALE
5 PLIES
11 PLIES
.172 REF
6 PLIES
.091
REF
MILLED
-29 REF
-9 REF
-7 REF
-5 REF
DOUBLER
FOR PLY NUMBERS
& DROP OFFS SEE LINE
DOUBLER
FOR PLY DASH NUMBERS
& DROP OFFS SEE LINE
DIAGRAM V & TABLE 4
VIEW N
SCALE 1/1
BL 0.000
WL 129.200
SECTION J-J
STA 496.280
-3 REF
7-31111 4737 INSERT
POT INSERT IN CORE
2.25 DIA THRU WITH
FOAM ADHESIVE
-29 REF
-33 REF
WL 129.200 TRACE
VIEW L-L
SCALE 1/1

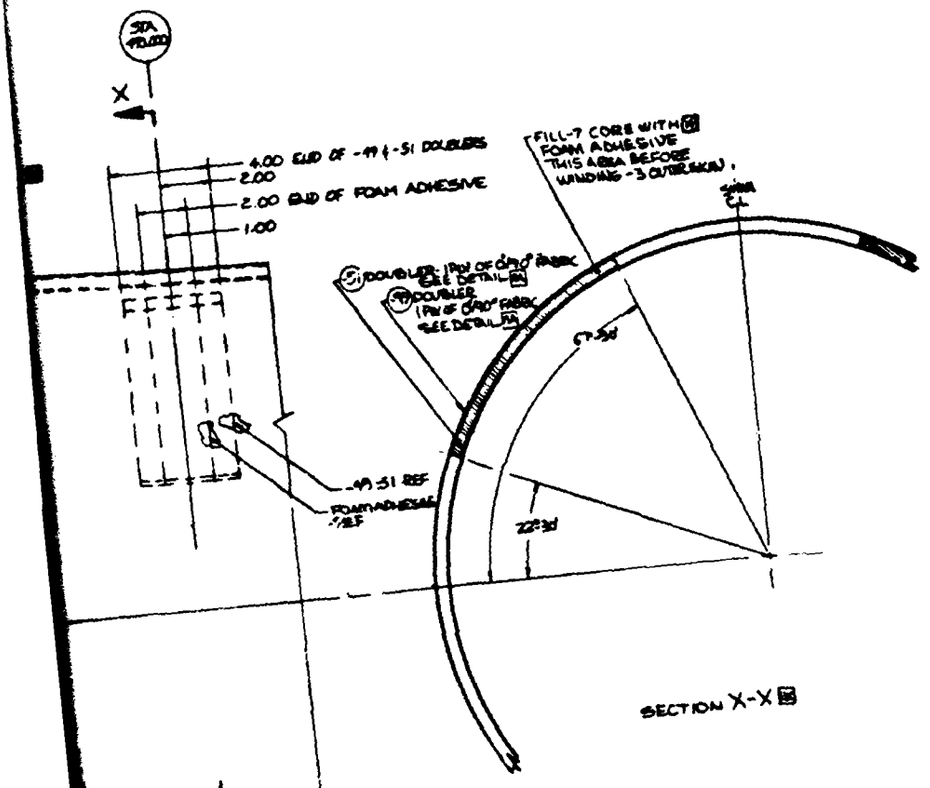


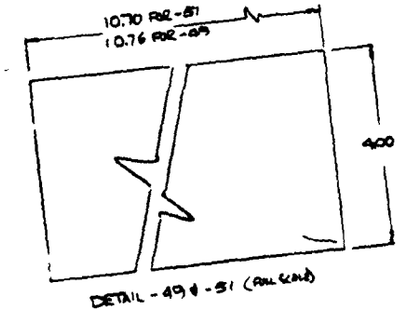
DETAIL A
SCALE: 1/1






EOP -31, -37, -41
BARRIER -37
7 REQD - INSERT -21
SEE DETAIL
.198-.204 DIA THRU 7 PL
TO MATCH 7-2H350012
EOP -31, -37, -41
-45 REF
4.90
1.10
-43 REF
5.00
1.40
5.20
3.650
1.950
2.00
1.000
2.000
.015
1.10
5.50
1.20
3.90
EOP -31, -37, -41
VIEW Y-Y
(FULL SIZE)
STA 397.00
BL 0.000
STA 430.148
STA 450.000
STA 470.000
BARRIER -37
BARRIER -45
BARRIER
SECTION Q-Q
-9 REF
-3 REF
-7 REF
Q
Y
2.60
.60
X
DOUBLER 1 PLY OF 0°/90° FABRIC
DOUBLER 1 PLY OF 0°/90° FABRIC
3





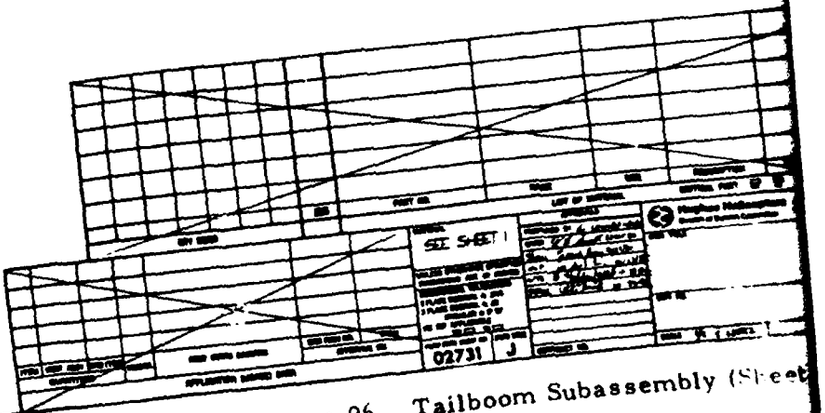
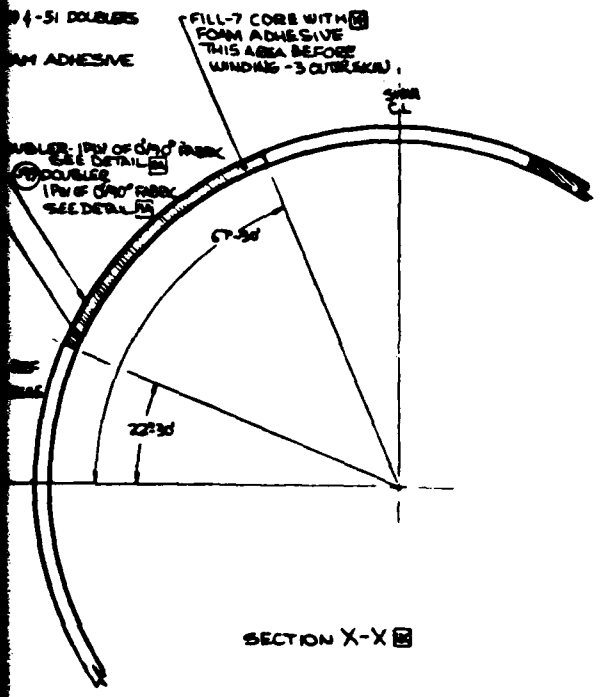


Figure 96. Tailboom Subassembly (Sheet

4



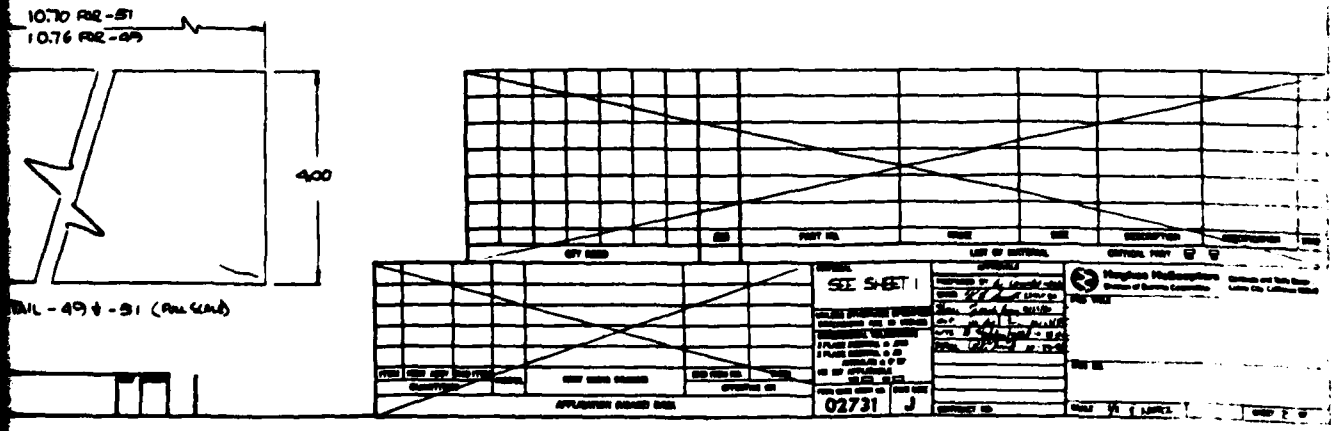
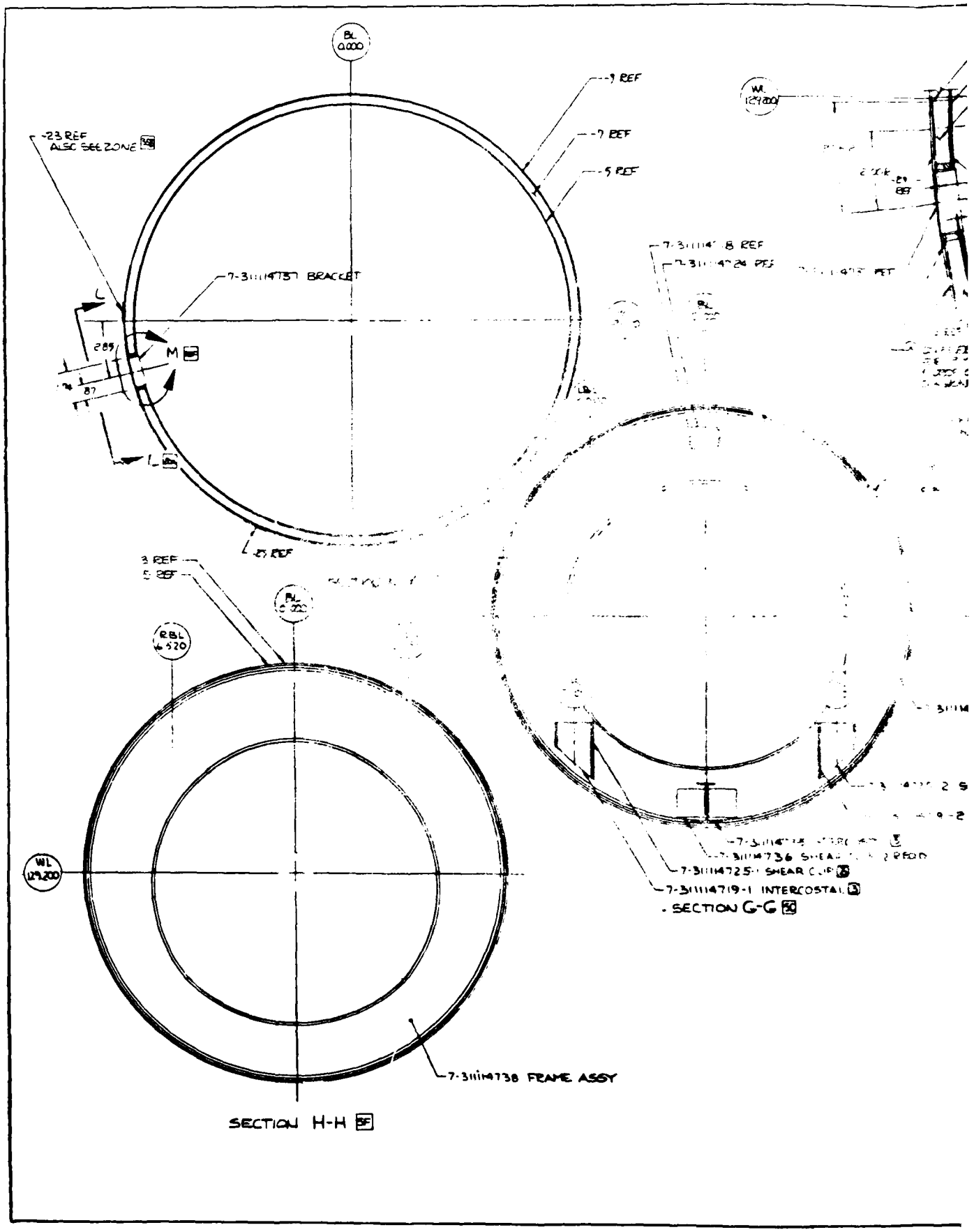
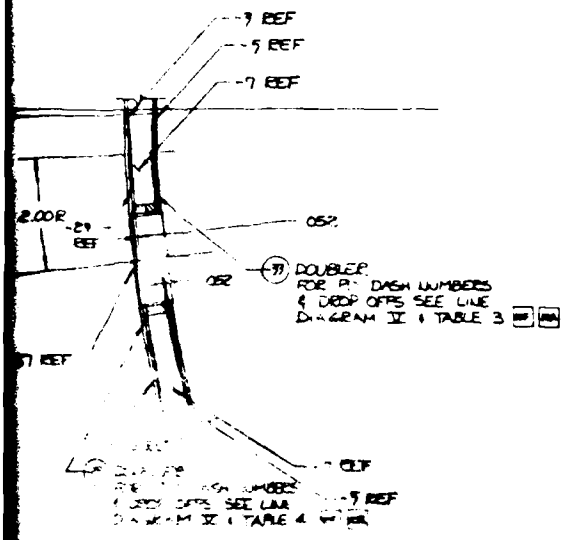
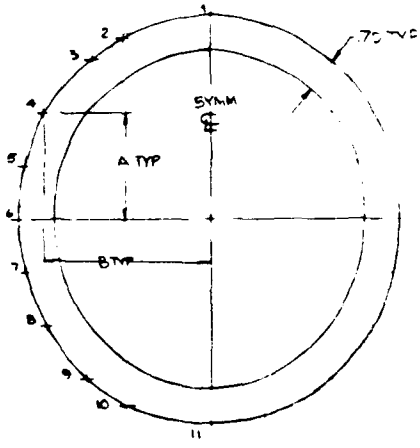


Figure 96. Tailboom Subassembly (Sheet 2 of 4)

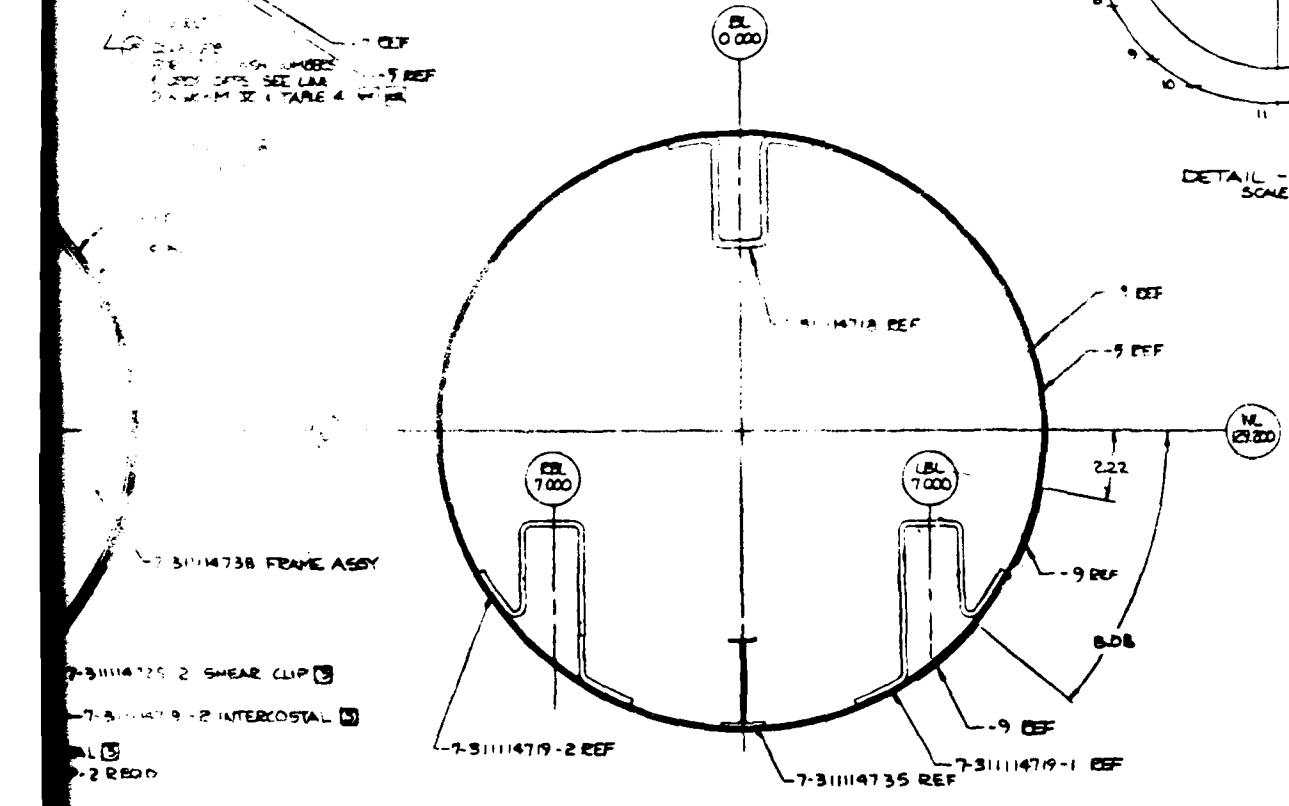

BL 0.000
-9 REF
-7 REF
-5 REF
-23 REF
ALSC SEE ZONE
7-3111147357 BRACKET
289
87
M
L
WL 129.200
7-3111147 8 REF
BL
3 REF
5 REF
RBL 6.520
7-3111147 36 SHEAR
7-3111147 25-1 SHEAR CLIP
7-3111147 19-1 INTERCOSTAL
SECTION G-G
7-3111147 38 FRAME ASSY
SECTION H-H



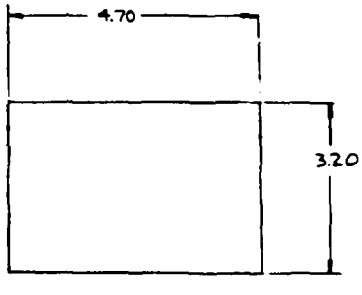


DIMENSION		
	A	B
1	3.80	6
2	3.50	40
3	3.00	2.20
4	2.00	3.08
5	.00	3.84
6	0	3.66
7	1.00	3.44
8	2.00	3.08
9	3.00	2.32
10	3.50	.64
11	3.80	0

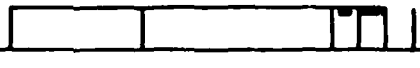
DETAIL -29
SCALE 1/1



SECTION E-E
SECTION F-F



DETAIL -13 SHT 1
SCALE 1/1



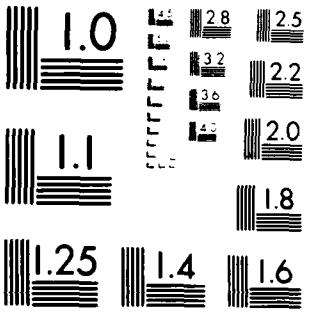
AD-A121 172 MANUFACTURING METHODS AND TECHNOLOGY (MANTECH) PROGRAM MANUFACTURING TECH.'.(U) HUGHES HELICOPTERS INC CULVER CITY CA J V ALEXANDER ET AL. OCT 81 HHI-81-367 3/3

UNCLASSIFIED USAAVRADCOM-TR-82-F-1 DAAK50-78-G-0004 F/G 1/3 NL

END

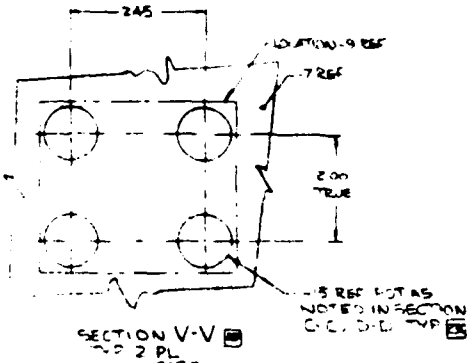
FILMED

DTIC

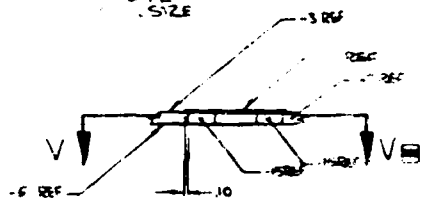


.70 TYP

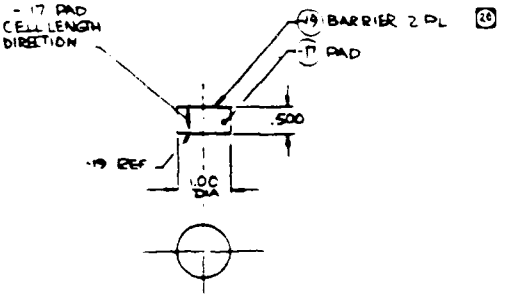
	DIMENSION	
	A	B
1	3.80	⊖
2	3.50	1.40
3	3.00	2.20
4	2.00	3.08
5	1.00	3.84
6	⊖	3.66
7	1.00	3.44
8	2.00	3.08
9	3.00	2.32
10	3.50	1.64
11	3.80	⊖



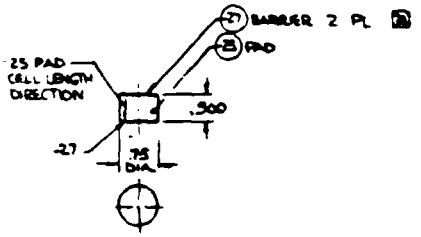
SECTION V-V
2 PL
SIZE



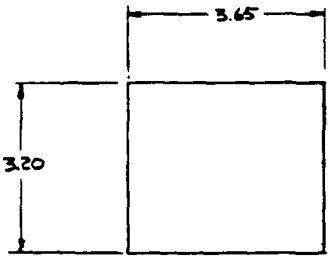
SECTION N-N



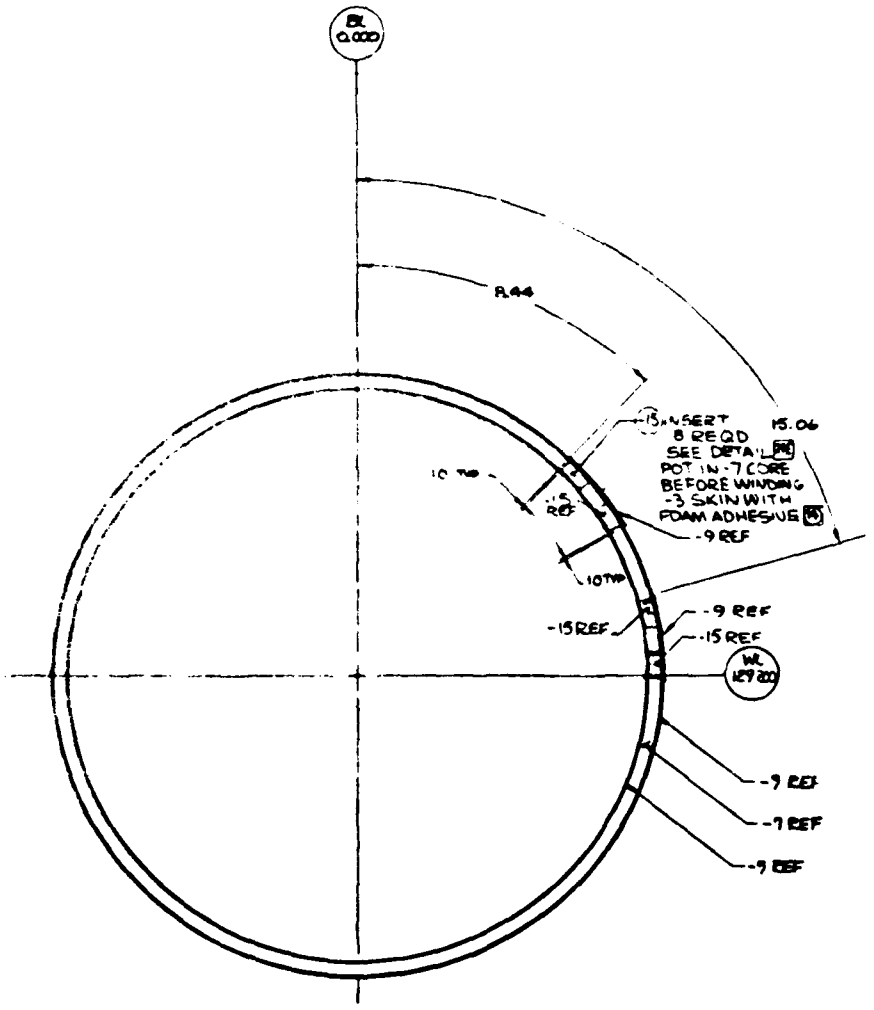
DETAIL -15 INSERT
FULL SIZE



DETAIL-21 INSERT
FULL SIZE



DETAIL -9
SCALE: 1/1



SECTION C-C
SECTION D-D

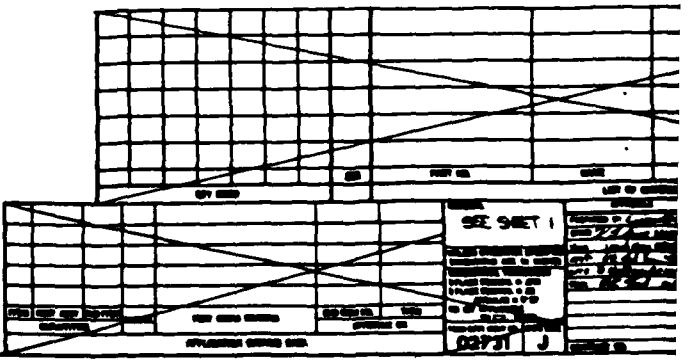


Figure 96. Tailboom Subas

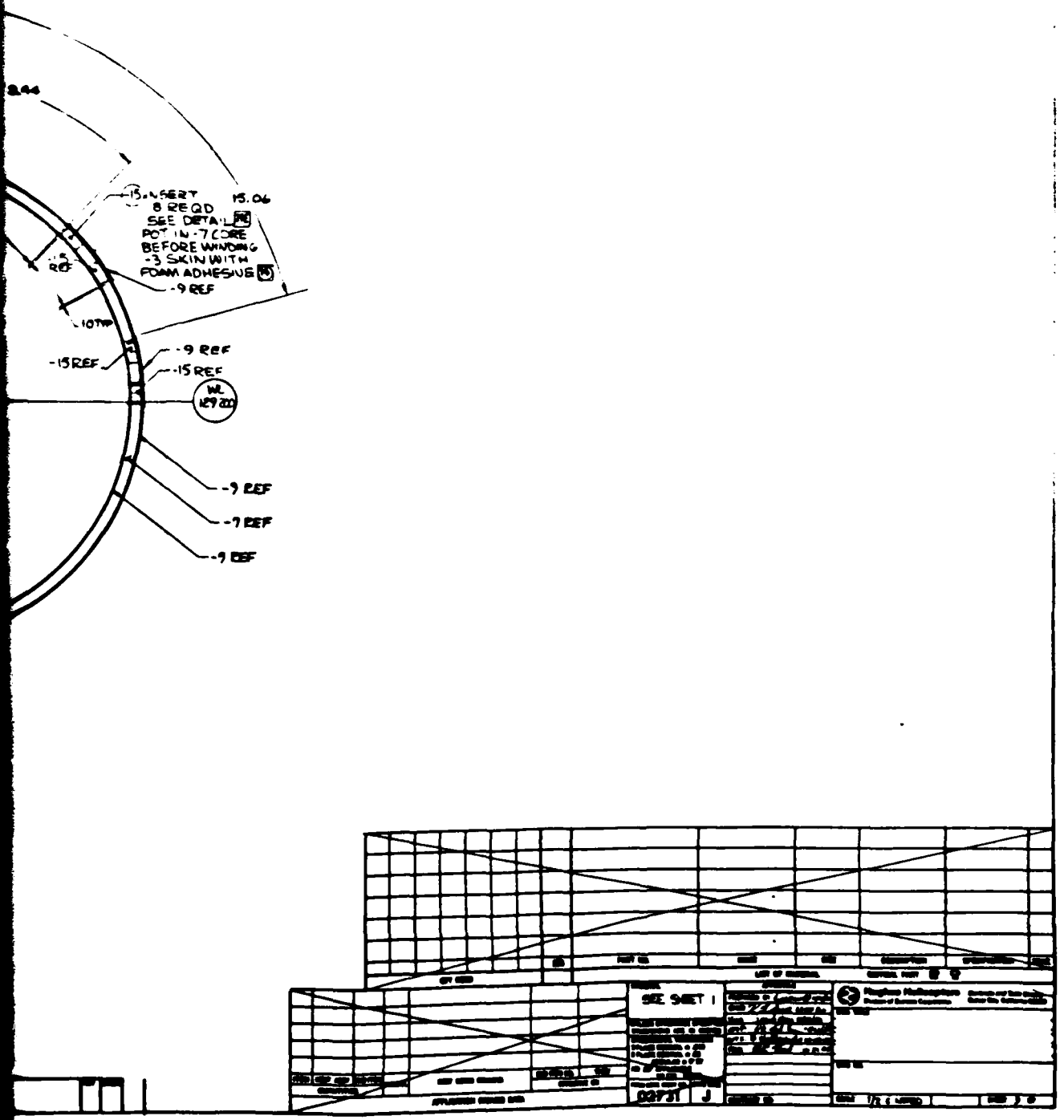
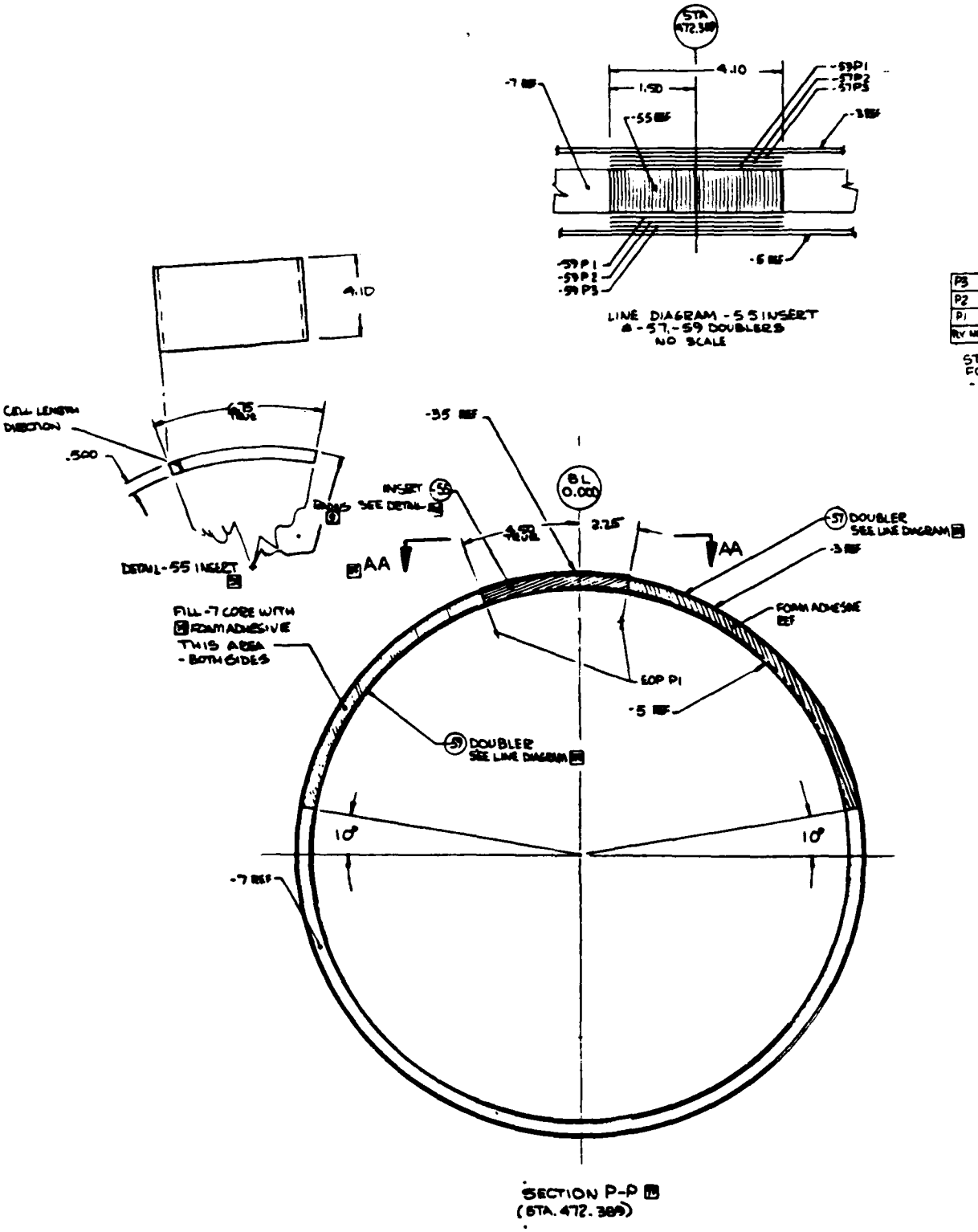
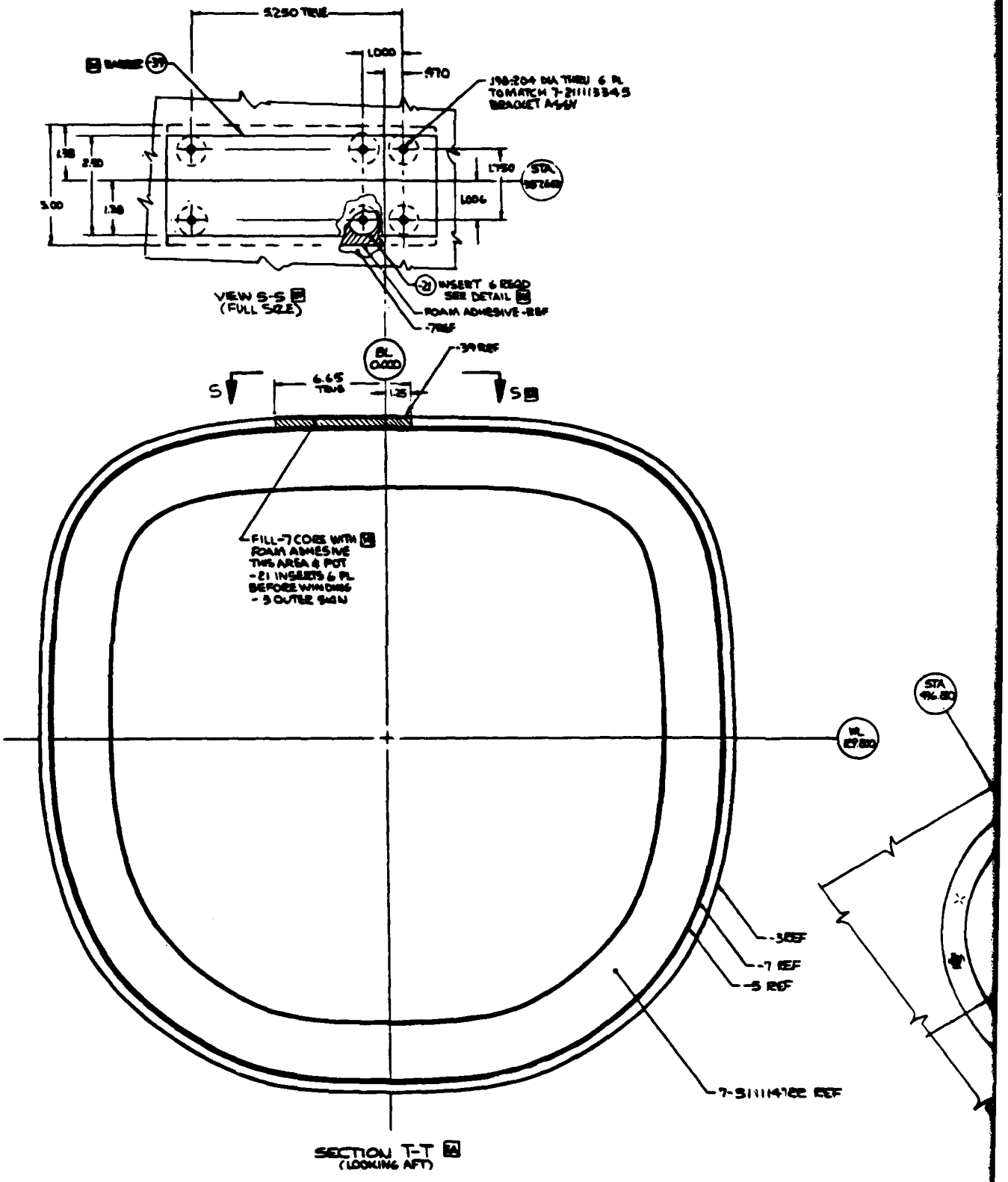


Figure 96. Tailboom Subassembly (Sheet 3 of 4)



P3	145°			
P2	90°			
P1	0/90°			
REV NO	ORIENT	MAT'L		

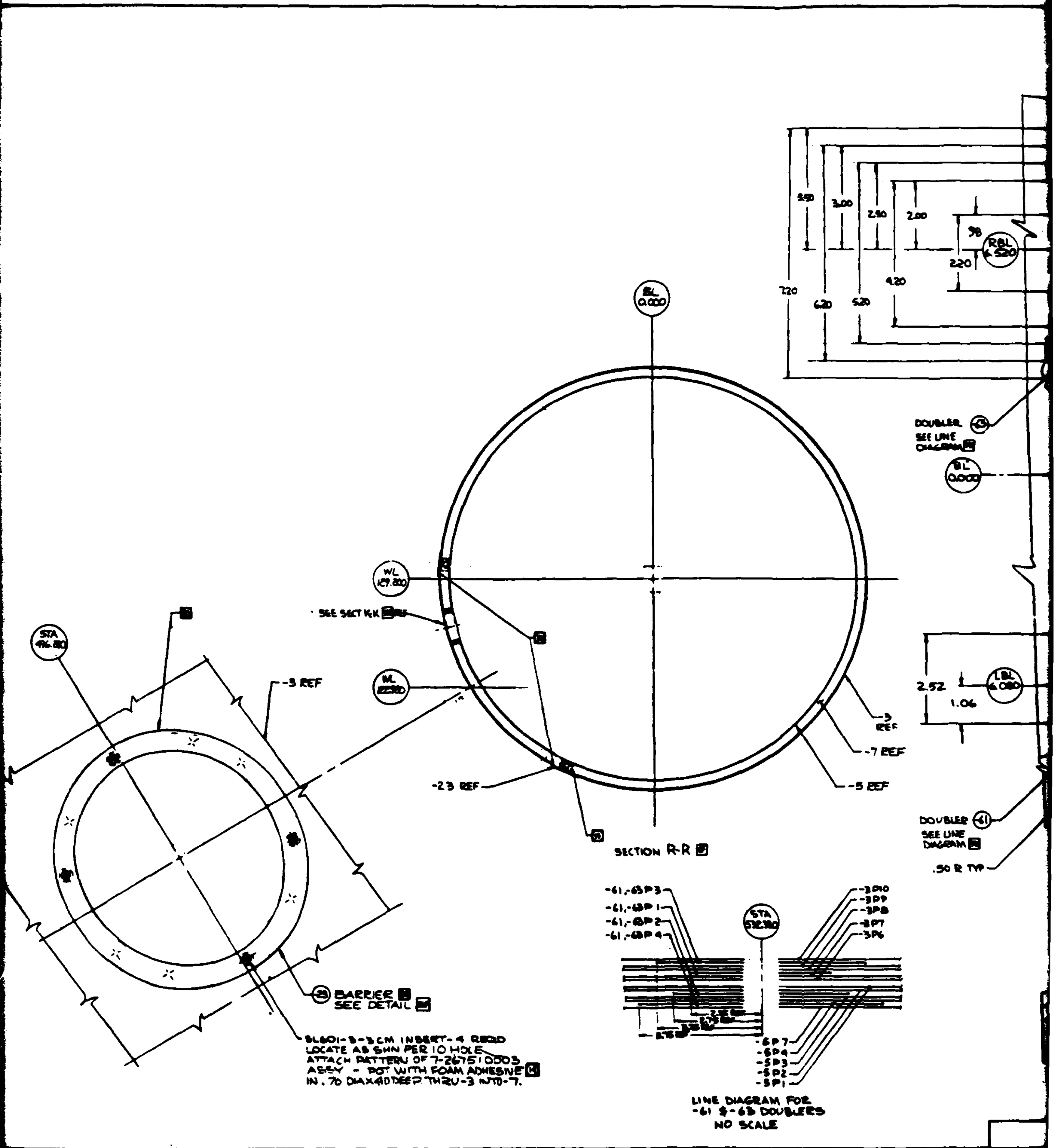
STACKING SEQUENCE FOR -55 INSERT -57, -59 DOUBLERS



	145°			.017
	90°			.030
	d/90°			.010
NO	ORIENT	MATL		PLY THK

STACKING SEQUENCE
FOR -55 INSERT @
-57.-59 DOUBLERS

2



SECTION R-R
-3 REF
-7 REF
-5 REF
-23 REF
BARRIER
SEE DETAIL
DOUBLER
SEE LINE DIAGRAM
.50 R TYP
2.52
1.06
3.50
3.00
2.50
2.00
7.20
6.20
5.20
4.20
2.20
LINE DIAGRAM FOR -61 & -63 DOUBLERS NO SCALE

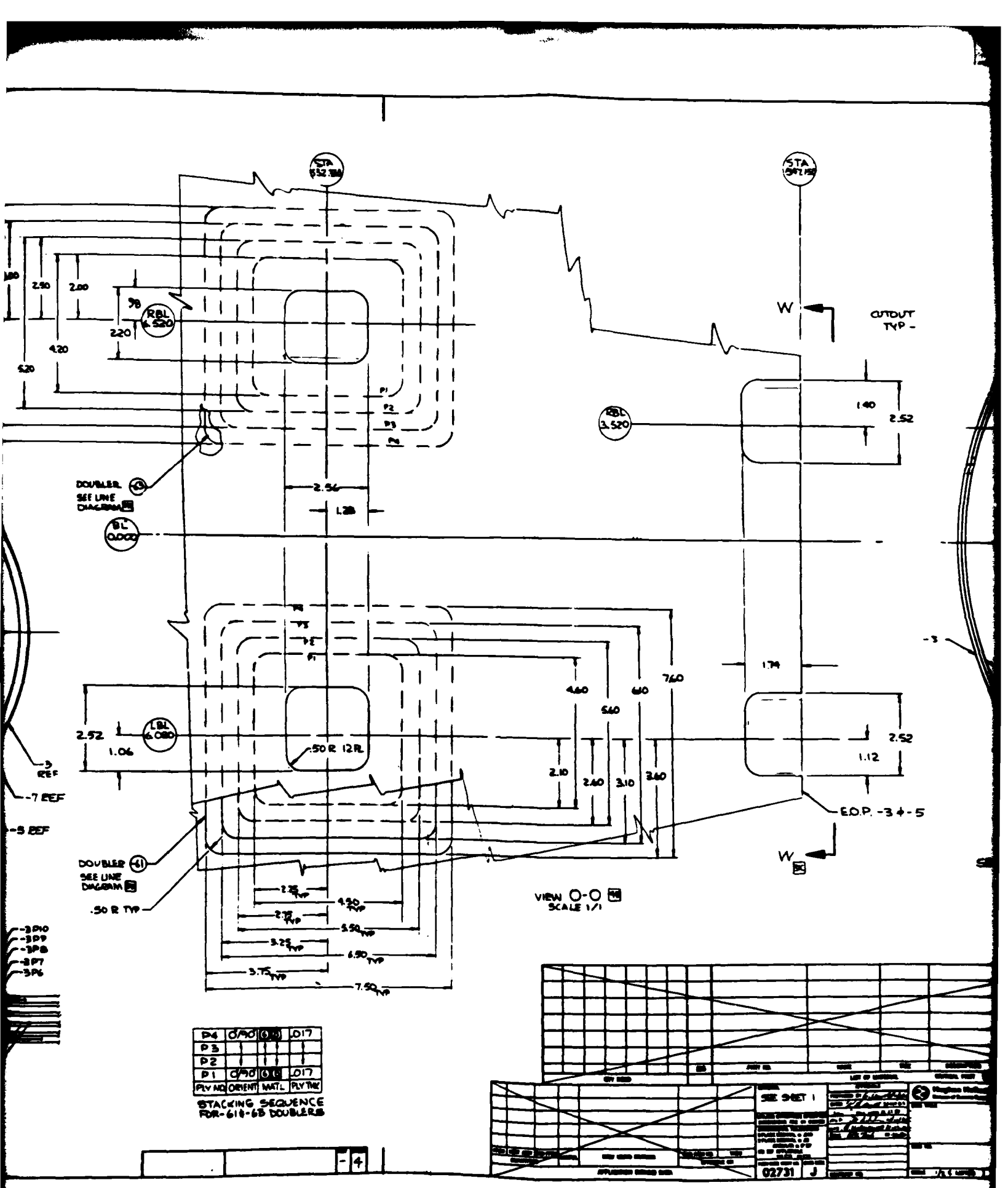


Figure 96. Tailboom Subassembly (She

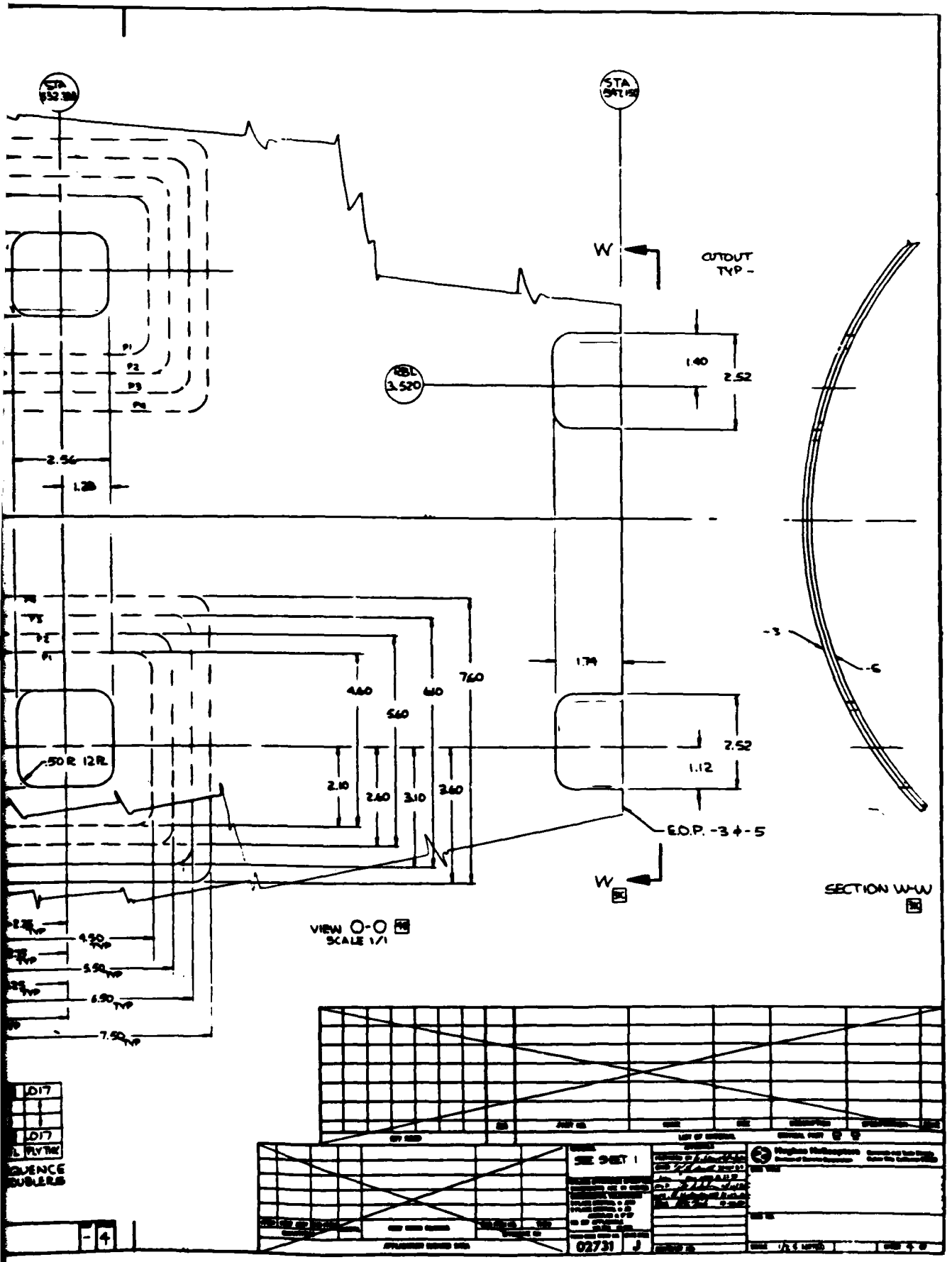
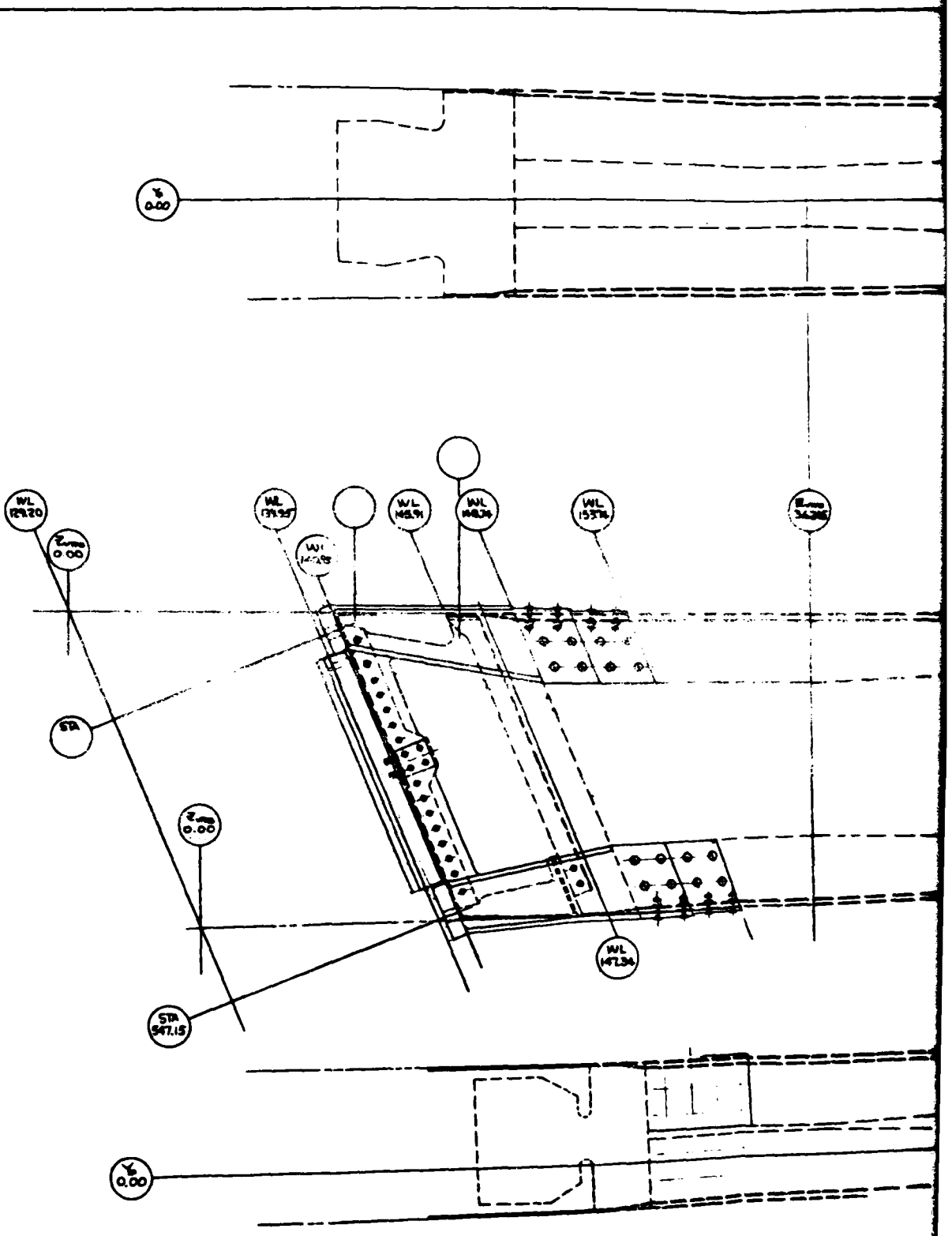
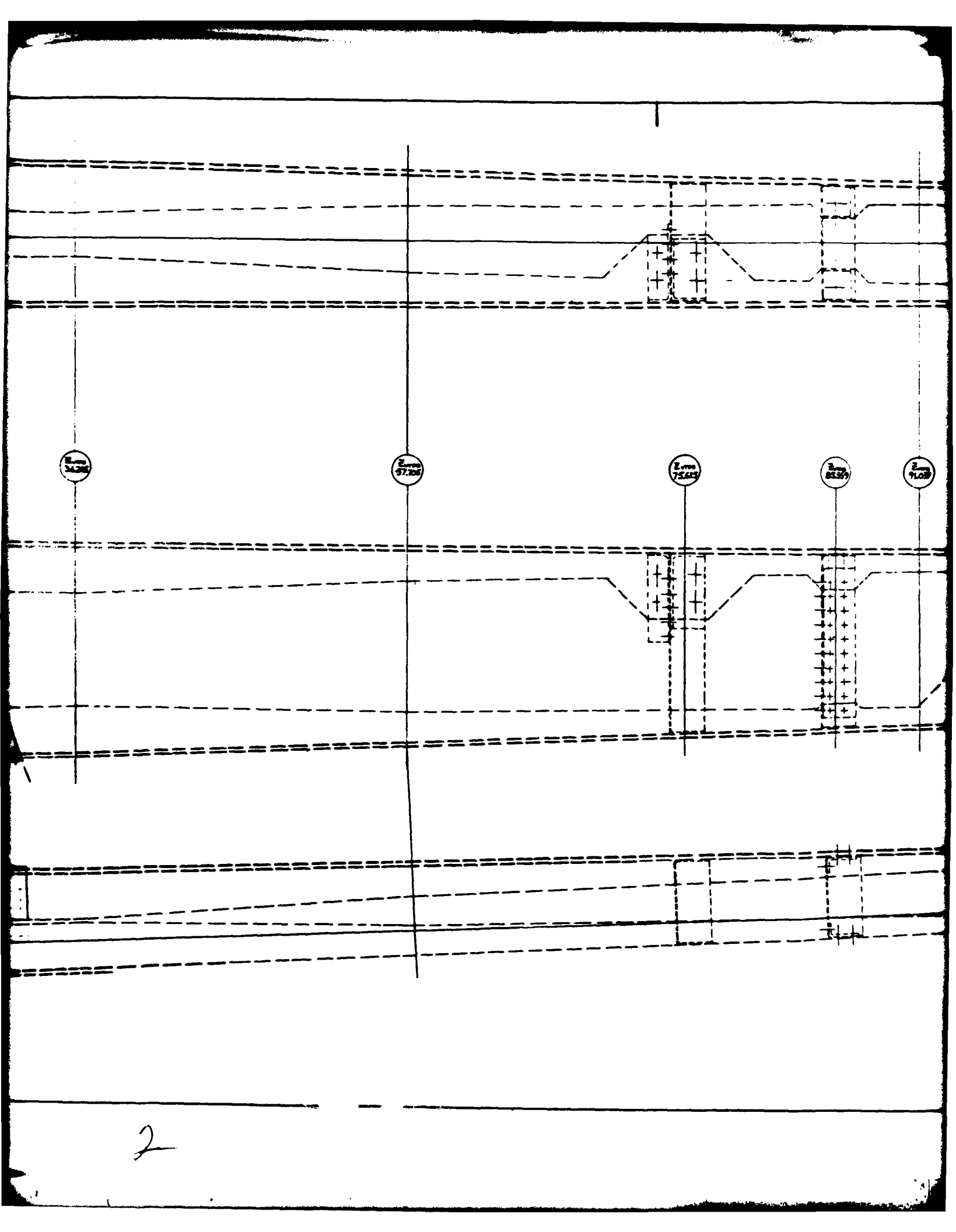
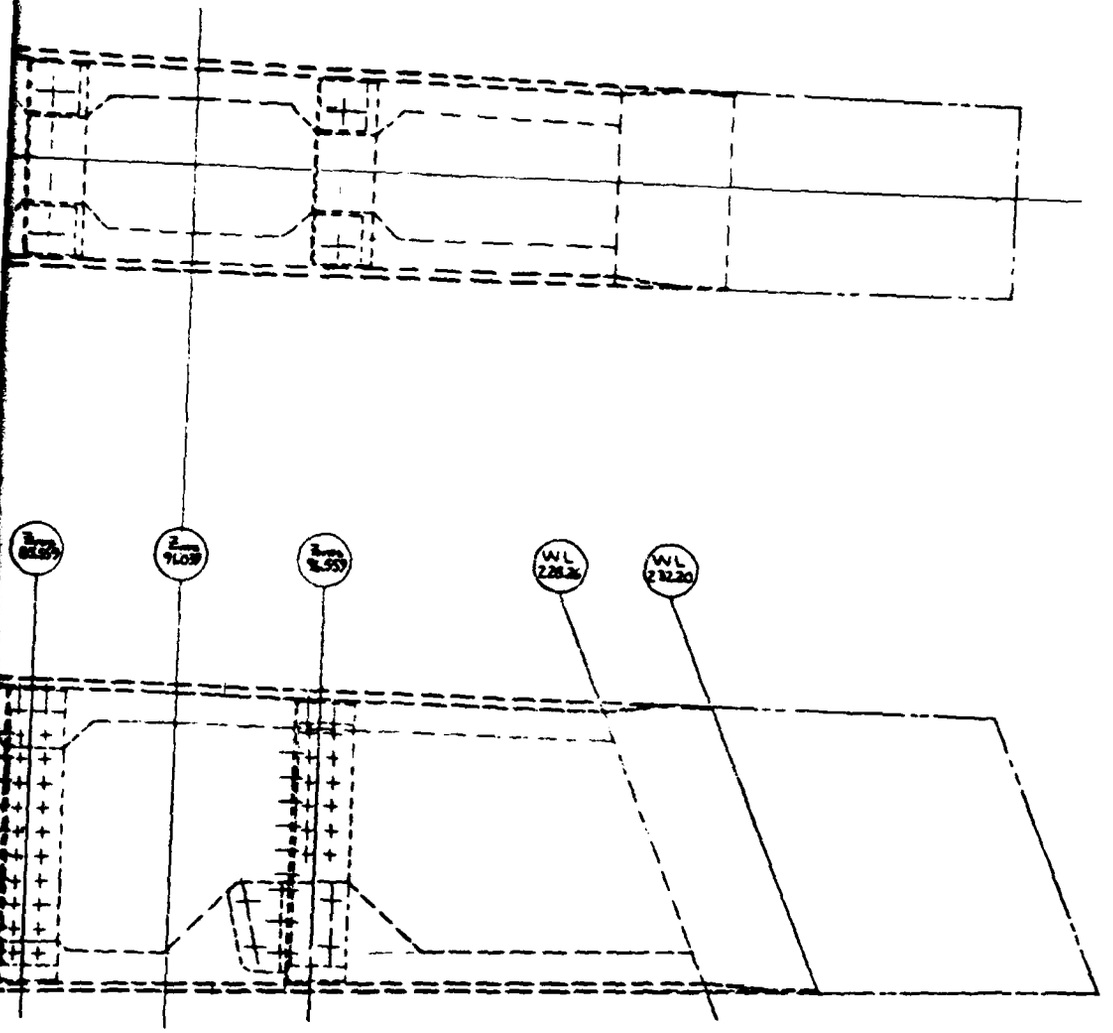


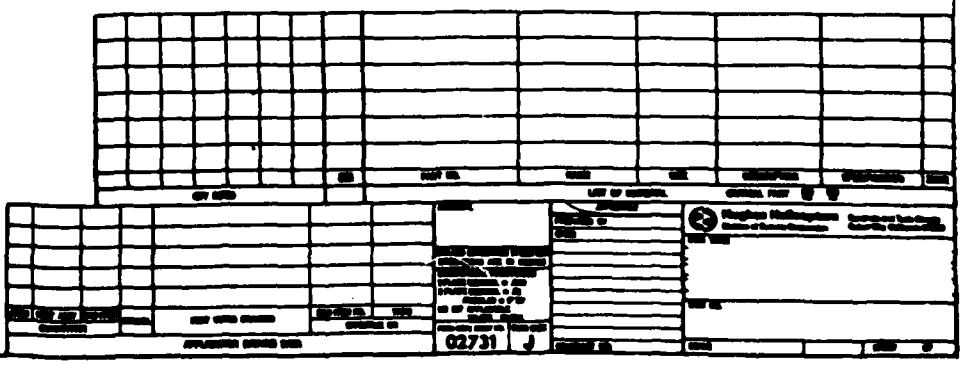
Figure 96. Tailboom Subassembly (Sheet 4 of 4)


WL 129.20
WL 139.95
WL 146.91
WL 148.36
WL 155.16
WL 147.34
STA 547.15
0.00
0.00
0.00



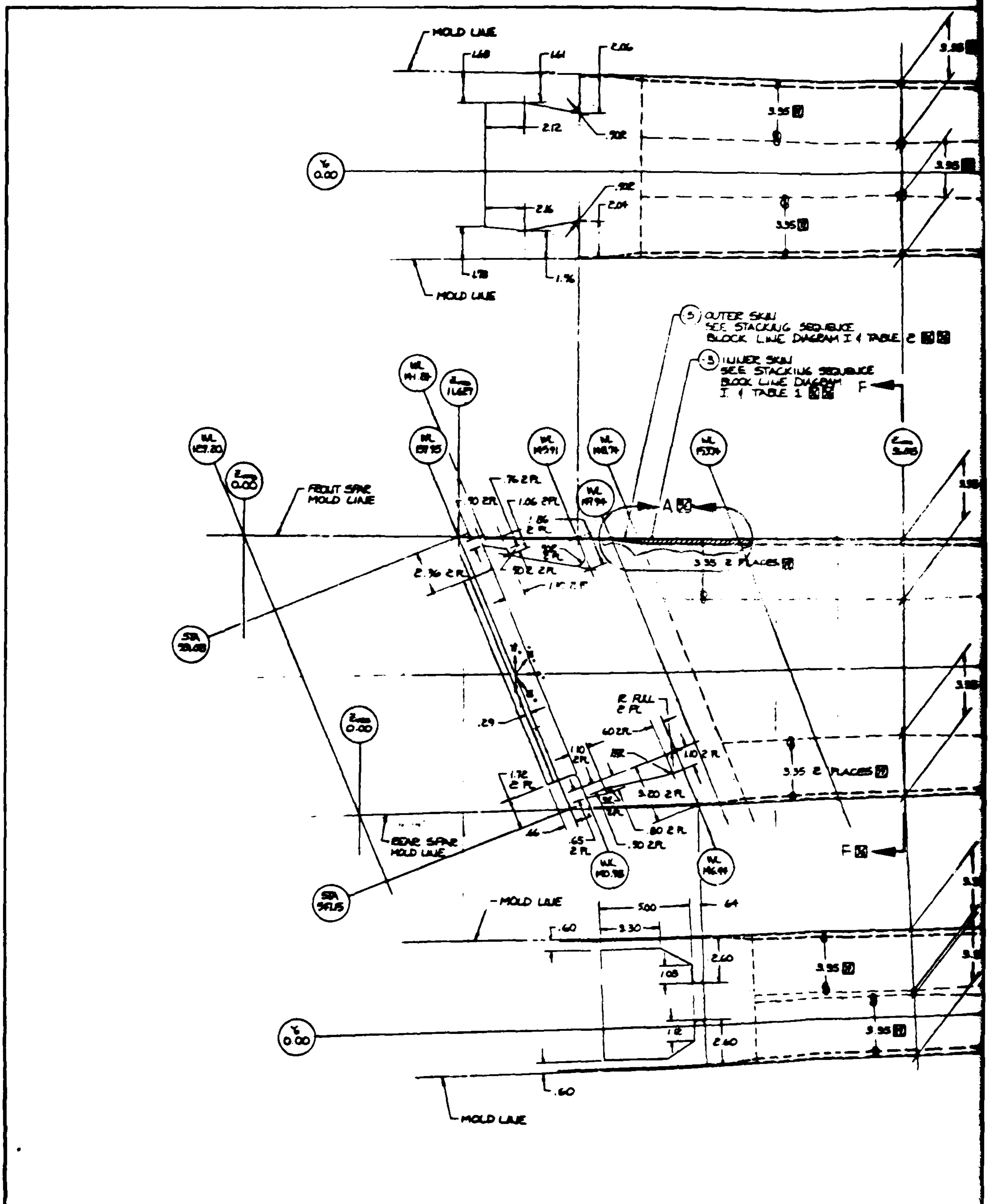
2

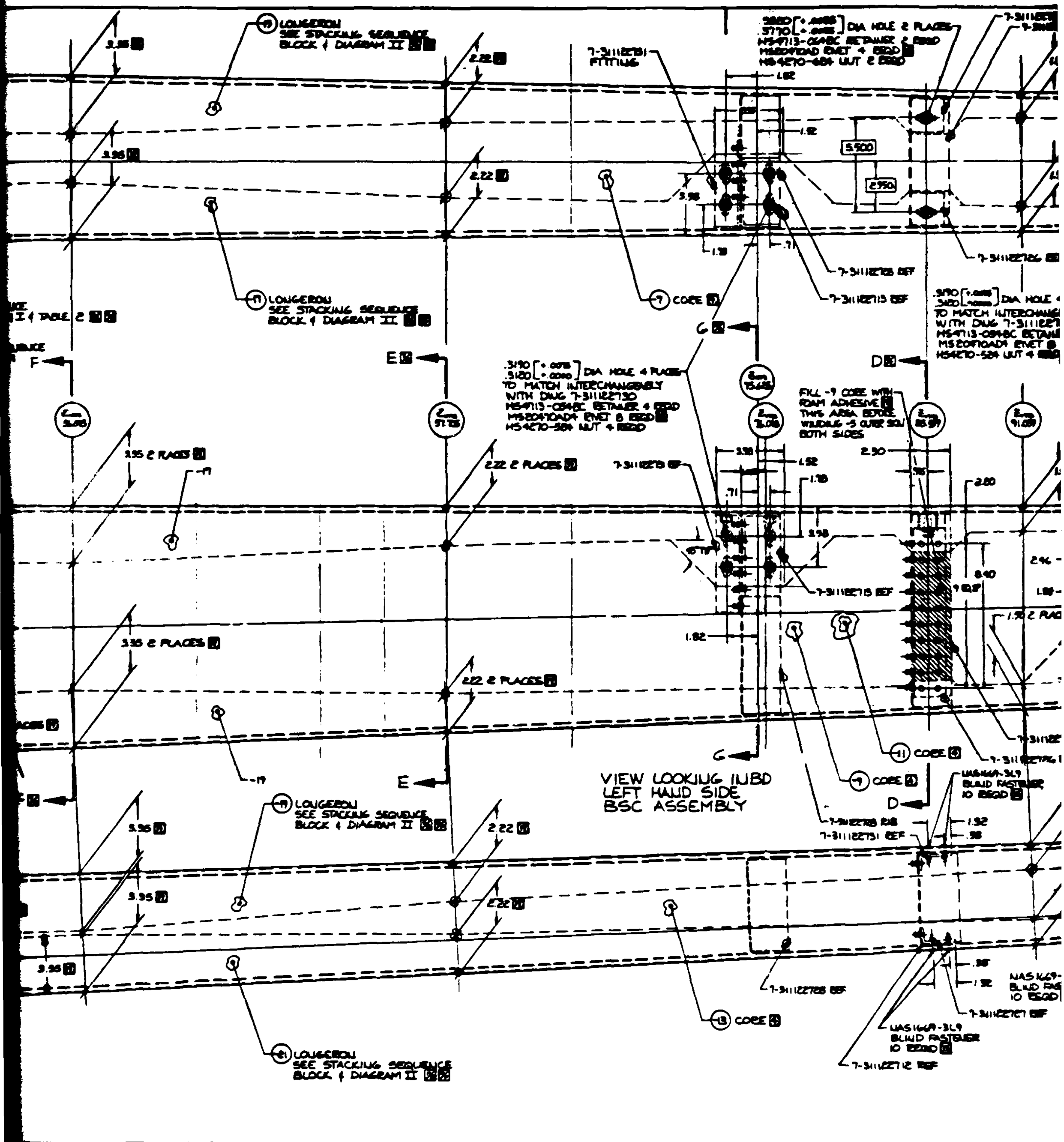

WL
WL



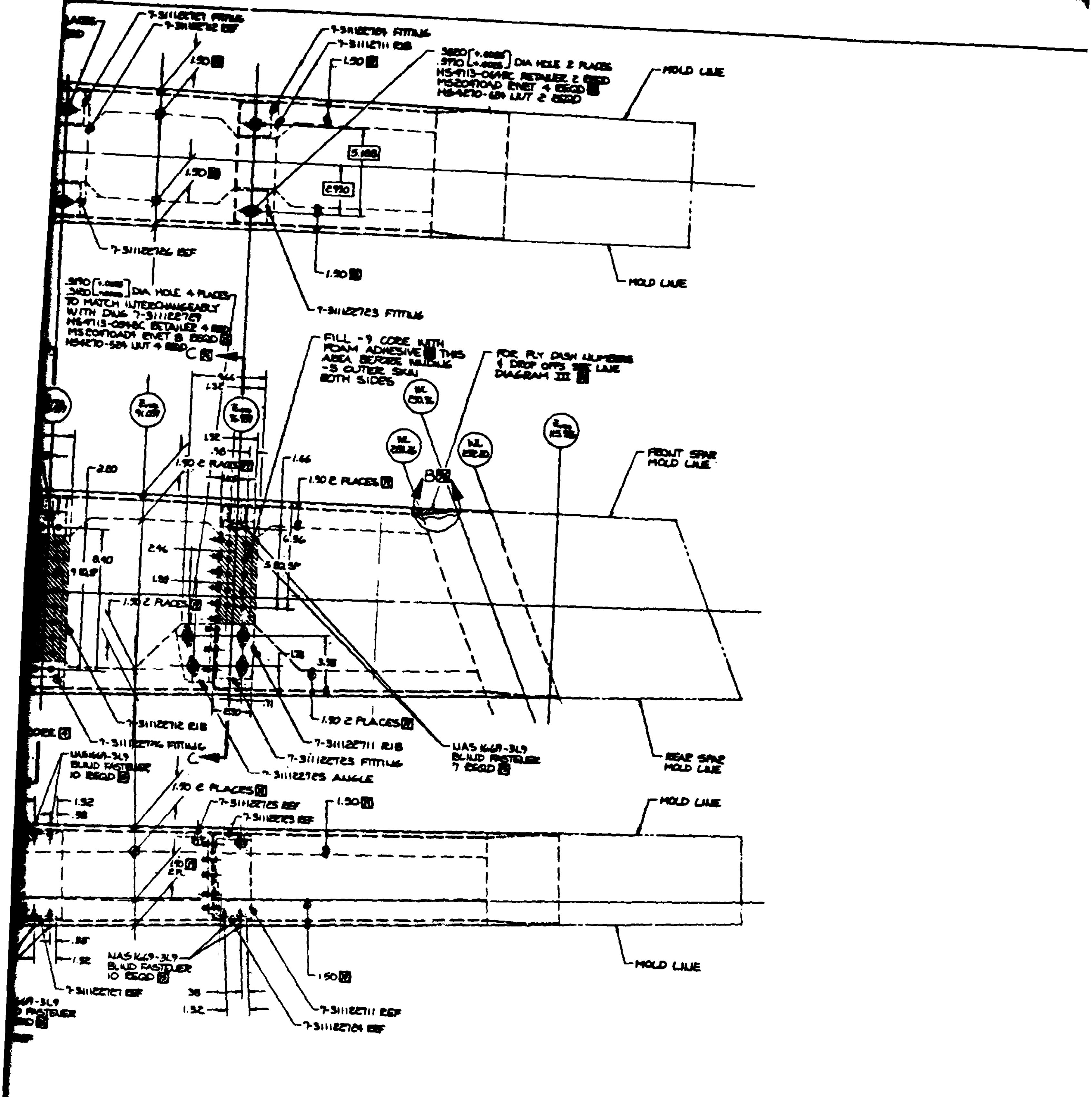
4

Figure 97. Vertical Tail Assembly


MOLD LINE
OUTER SKIN
SEE STACKING SEQUENCE
BLOCK LINE DIAGRAM I & TABLE 2
INNER SKIN
SEE STACKING SEQUENCE
BLOCK LINE DIAGRAM
I & TABLE 1
FRONT SPAR
MOLD LINE
REAR SPAR
MOLD LINE
3.35 2 PLACES
R FULL
2 PL



LONGERON SEE STACKING SEQUENCE BLOCK & DIAGRAM II
7-311122781 FITTING
VIEW LOOKING INBD
LEFT HAND SIDE
BSC ASSEMBLY
CORE
BLIND FASTENER 10 REQD
FILL -9 CORE WITH FOAM ADHESIVE THIS AREA BEFORE WINDING -5 OUTER SKIN BOTH SIDES


7-3111122727 FITTING
7-3111122712 REF
7-3111122724 FITTING
7-3111122711 RIB
1.50
.3800 .3770 DIA HOLE 2 PLACES
MOLD LINE
5.188
2.930
7-3111122726 REF
MOLD LINE
7-3111122723 FITTING
DIA HOLE 4 PLACES TO MATCH INTERCHANGEABLY WITH DWG 7-3111122727
FILL -9 CORE WITH FOAM ADHESIVE THIS AREA BEFORE INSTALLING -5 OUTER SKIN BOTH SIDES
FOR PLY DASH NUMBERS & DROP OFFS SEE LINE DIAGRAM III
FRONT SPAR MOLD LINE
1.90 2 PLACES
1.90 2 PLACES
1.90 2 PLACES
1.90 2 PLACES
7-3111122712 RIB
7-3111122711 RIB
7-3111122723 FITTING
7-3111122725 ANGLE
NAS 1669-3L9 BLIND FASTENER 7 REQD
REAR SPAR MOLD LINE
NAS 1669-3L9 BLIND FASTENER 10 REQD
1.90 2 PLACES
7-3111122725 REF
MOLD LINE
MOLD LINE
NAS 1669-3L9 BLIND FASTENER 10 REQD
7-3111122727 REF
7-3111122711 REF
7-3111122724 REF

3

AR		MILLED GLASS FIBRES		[11]
AR	HMS 16-1111, CL 3	FILM ADHESIVE		
AR	HMS 17-1002	SYNTACTIC FOAM		
74	UAS1669-3L9	BLIND FASTENER		
80	HLT5G-6AGN	COLLAR		
4	HL48-6-5	PIN		
24	HL48-6-4	PIN		
52	HL48-6-3	PIN		
52	MS20470AD4	RIVET		
4	HS4713-0618C	RETAINER		
12	HS4713-0518C	RETAINER		
4	HS4210-624	NUT		
12	HS4210-524	NUT		
1	7-311122731	FITTING		
1	7-311122728	RIB		
1	7-311122727	FITTING		
1	7-311122726	FITTING		
1	7-311122725	ANGLE		
1	7-311122724	FITTING		
1	7-311122723	FITTING		
1	7-311122713	STRUT FITTING		
1	7-311122712	RIB		
1	7-311122711	RIB		
1	-21	LONGERON		[5] [6]
1	-19	LONGERON		[5] [6]
1	-17	LONGERON		[5] [6]
1	-15	LONGERON		[5] [6]
1	-13	CORE		[7]
1	-11	CORE		[7]
1	-9	CORE		[7]
1	-7	CORE		[7]
1	-5	OUTER SKIN	AR	[14] [6]
1	-3	INNER SKIN	AR	[13] [6]

21. INSTALL RIVETS PER HP 3-0 & HP 3-1
20. INSTALL RETAINERS PER HP 15-14
[19] DIMENSION SHOWN IS TO THE CENTER LINE OF THE EDGE OF THE CORE & LONGERON THEN SCARFED AT A 45° ANGLE AS SHOWN TYPICALLY IN SECTION C-C.
[18] SYNTACTIC FOAM PER HMS 17-1002
17. INSTALL BLIND FASTENERS PER HP 3-12
16. INSTALL HI-LOK FASTENERS PER HP 3-2
[15] SEAL FASTENERS USING HMS 16-1077 PER HP 3-10 TYPE IV
[14] 50% VOLUME GRAPHITE [5] ROVING & 50% VOLUME KEVLAR [13] ROVING (.017 TOTAL PLY THK)
[13] KEVLAR ROVING HMS 16-1164 TYPE I, CL 3
[12] HMS 16-1111 FILM ADHESIVE CL 3
[11] MILLED GLASS FIBRES/EPOXY 1/32 LENGTH THALCO, LA
[10] FINISH EXTERIOR ONLY CODE E PER EPB 4-230
9. IDENTIFY PER HP8-5 TYPE I, CL 3 LOC OPT
8. DIMENSIONING & TOLERANCING PER ANSI Y14.5
[7] CORE (NOMEX) HMS 16-1114 TYPE 6, CL 1
[6] EPOXY RESIN SYSTEM HMS 16-1115 TYPE I, CL 1
[5] GRAPHITE ROVING HMS 16-1163 TYPE I, CL 1 (.0085/PLY)
[4] FILM ADHESIVE BOND PER HP 16-30
3. FILAMENT WIND PER HP 15-67
[2] FIRST PLY GOES AGAINST TOOL FACE
1. FOR LOFT LINES SEE DWG. 7-311122700

NOTES: UNLESS OTHERWISE SPECIFIED

7-311122700

02731 J

Figure 98. Vertical Tail Subassembly (Shee

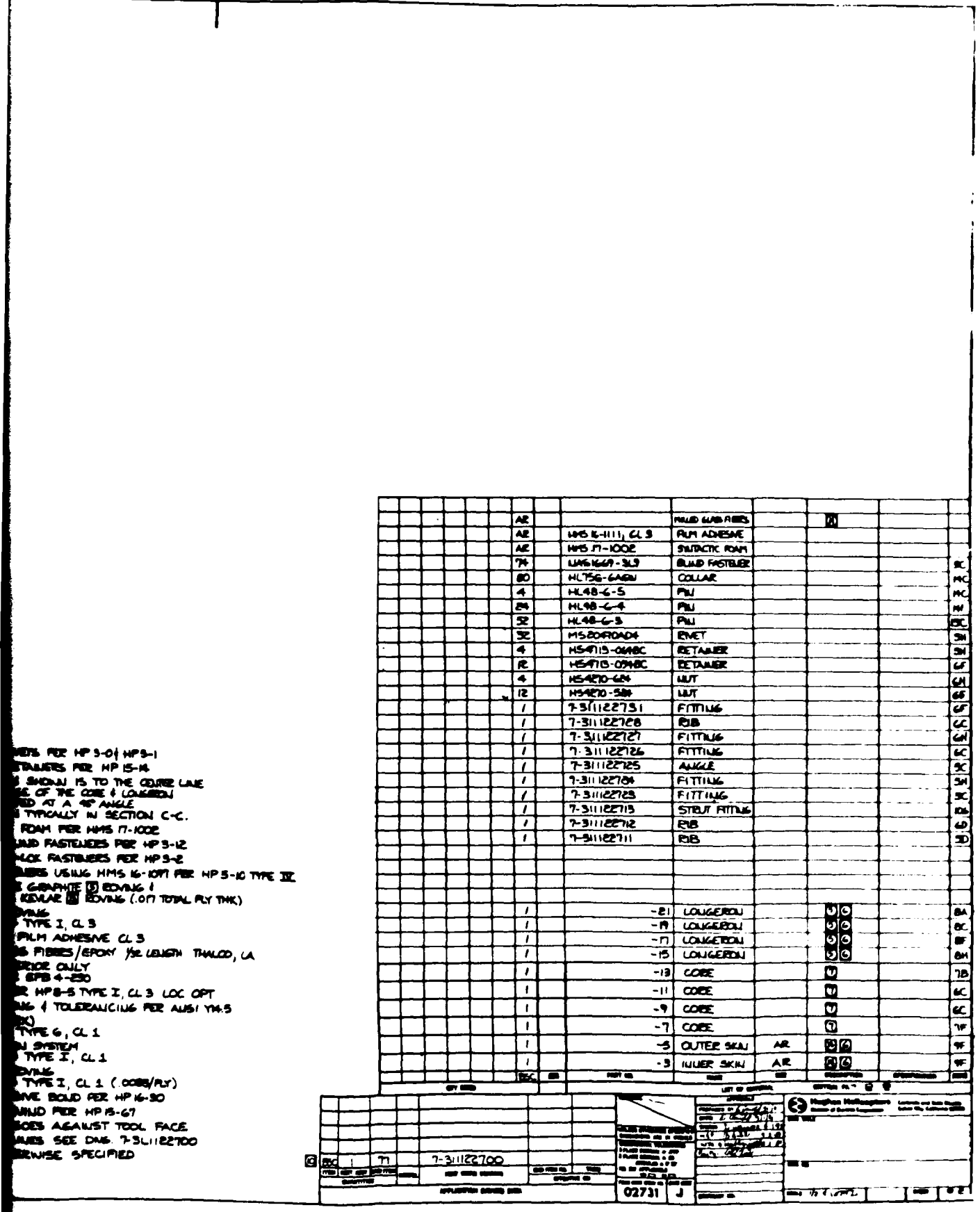
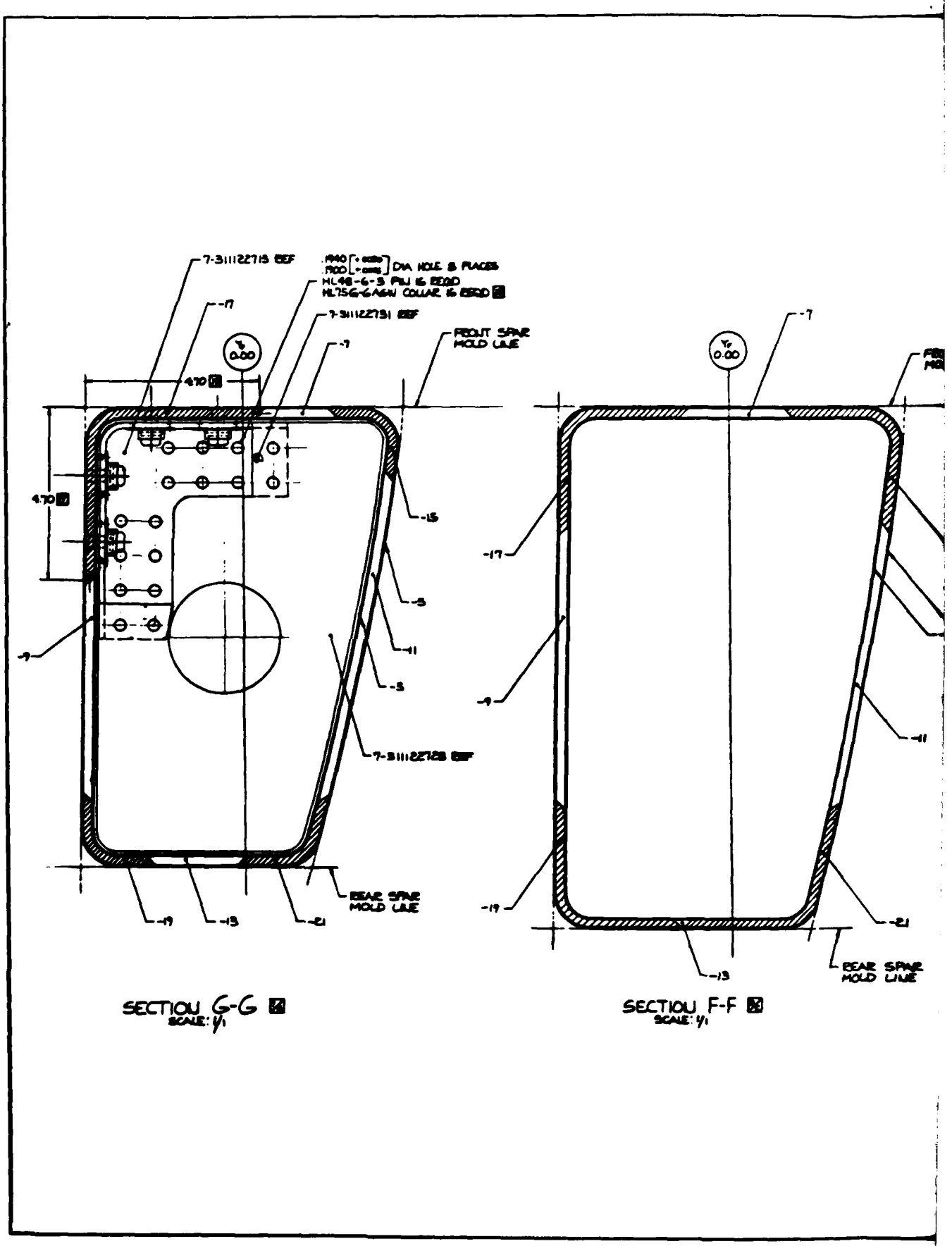
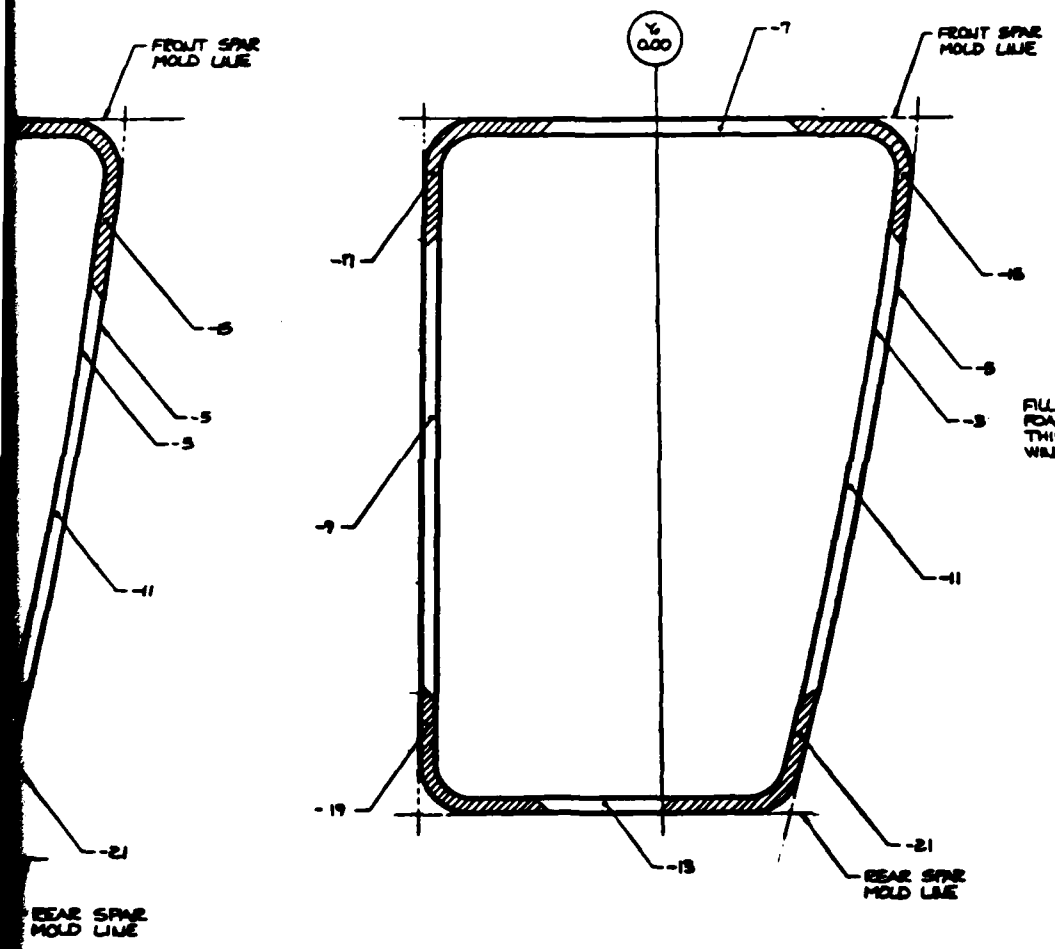
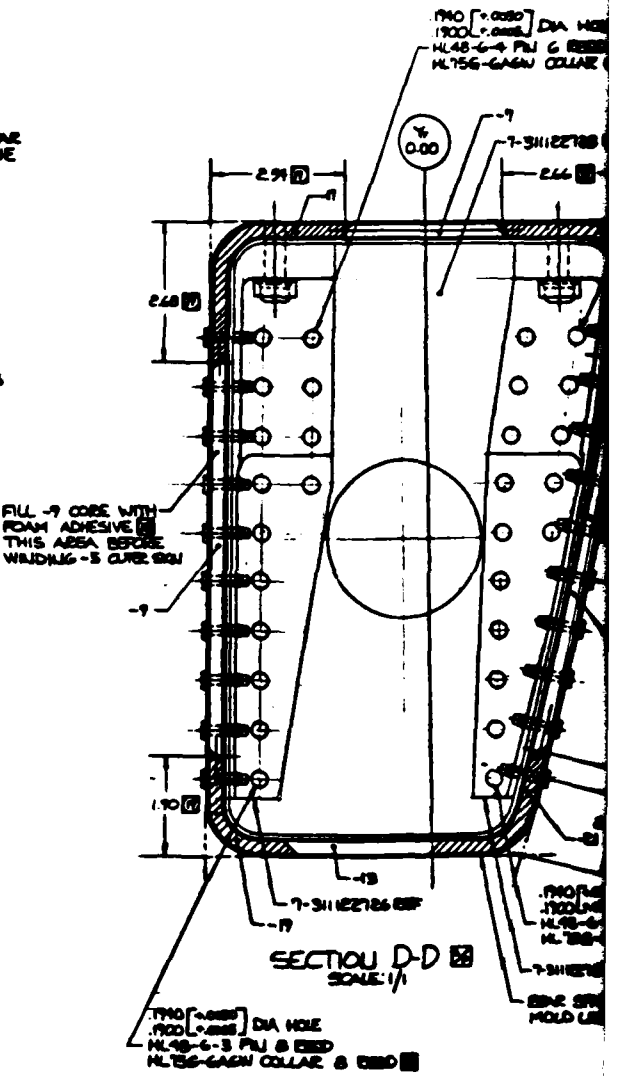


Figure 98. Vertical Tail Subassembly (Sheet 1 of 2)


7-311122713 REF
.1940 [+.0000 -.0005] DIA HOLE 8 PLACES
.1900 [+.0000 -.0005]
HL498-6-3 PIN IS REQD
HL75G-6 AGW COLLAR IS REQD
7-311122731 REF
-17
-7
FRONT SPAR MOLD LINE
0.00
4.70
4.70
-15
-5
-11
-9
-5
7-311122728 REF
REAR SPAR MOLD LINE
-19
-13
-21
SECTION G-G
SCALE: 1/1
-7
0.00
-17
-9
-11
-19
-21
-13
REAR SPAR MOLD LINE
SECTION F-F
SCALE: 1/1

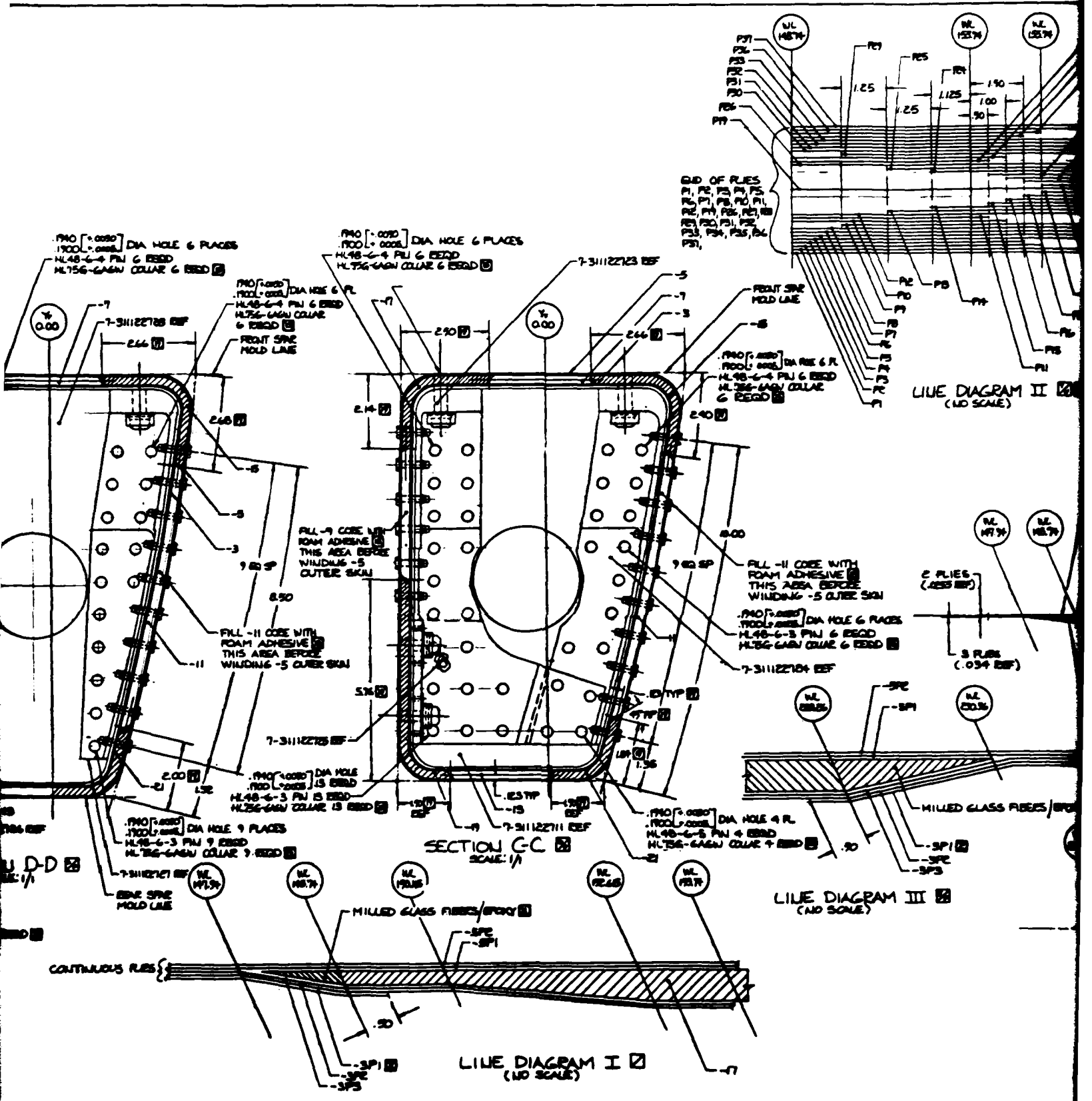


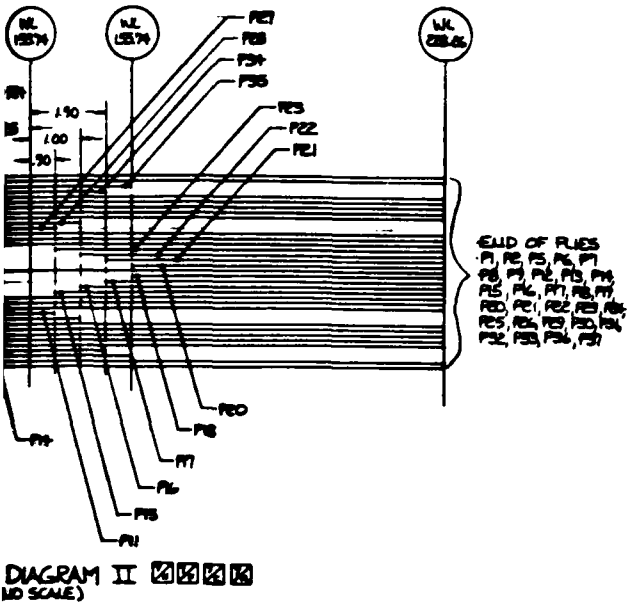
SECTION E-E
SCALE: 1/1

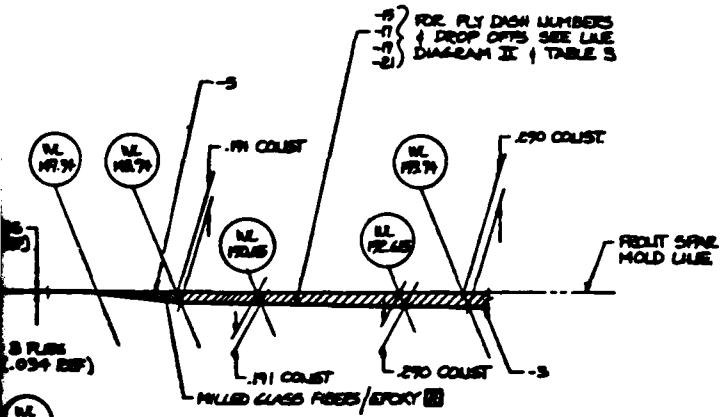


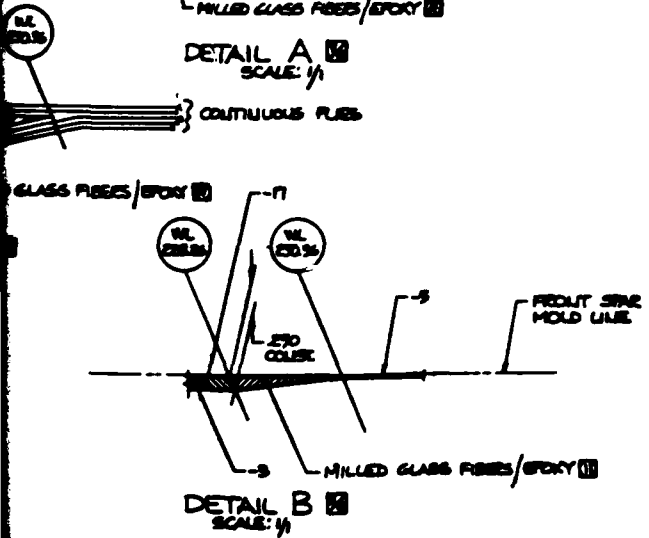
SECTION D-D
SCALE: 1/1

CONTINUOUS RUBB


LINE DIAGRAM II
(NO SCALE)
SECTION C-C
SCALE: 1/1
LINE DIAGRAM III
(NO SCALE)
LINE DIAGRAM I
(NO SCALE)
CONTINUOUS PLIES
MILLED GLASS FIBERS/EPOXY
FULL -11 CORE WITH FOAM ADHESIVE THIS AREA BEFORE WINDING -5 OUTER SKIN
FULL -9 CORE WITH FOAM ADHESIVE THIS AREA BEFORE WINDING -5 OUTER SKIN
FRONT SPAR MOLD LINE
REAR SPAR MOLD LINE
2 PLIES (.0255 REF)
3 PLIES (.039 REF)







P37	0°			.0085
P36	0°			
P35	+45°			
P34	-45°			
P33	0°			
P32	+45°			
P31	-45°			
P30	0°			
P29	0°			
P28	+45°			
P27	-45°			
P26	0°			
P25	0°			
P24	90°			
P23	0°			
P22	0°			
P21	0°			
P20	0°			
P19	90°			
P18	0°			
P17	0°			
P16	0°			
P15	0°			
P14	90°			
P13	0°			
P12	0°			
P11	-45°			
P10	+45°			
P9	0°			
P8	0°			
P7	-45°			
P6	+45°			
P5	0°			
P4	-45°			
P3	+45°			
P2	0°			
P1	0°			.0085
PLY NO.	ORIENT	MATL	PLY THK.	

STACKING SEQUENCE FOR -15,-17,-19,&-21 LOWER SKIN TABLE 3

P3	90°			.0085
P2	±38°			.071
P1	90°			.0085
PLY NO.	ORIENT	MATL	PLY THK.	

STACKING SEQUENCE FOR -3 INNER SKIN TABLE 1

P2	90°			.0085
P1	±38°			.071
PLY NO.	ORIENT	MATL	PLY THK.	

STACKING SEQUENCE FOR -5 OUTER SKIN TABLE 2

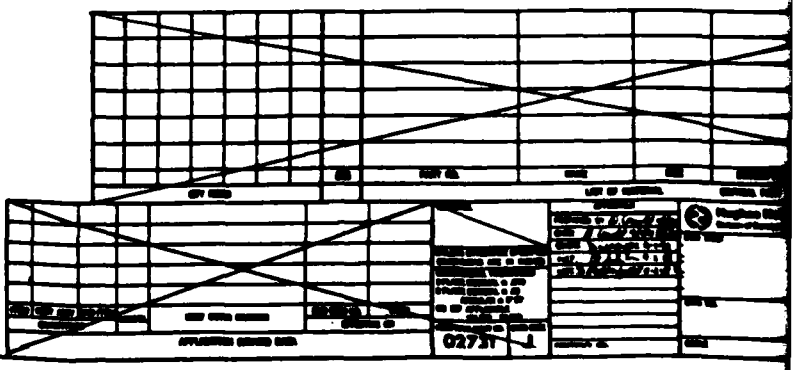


Figure 98. Vertical Tail Subassembly (S

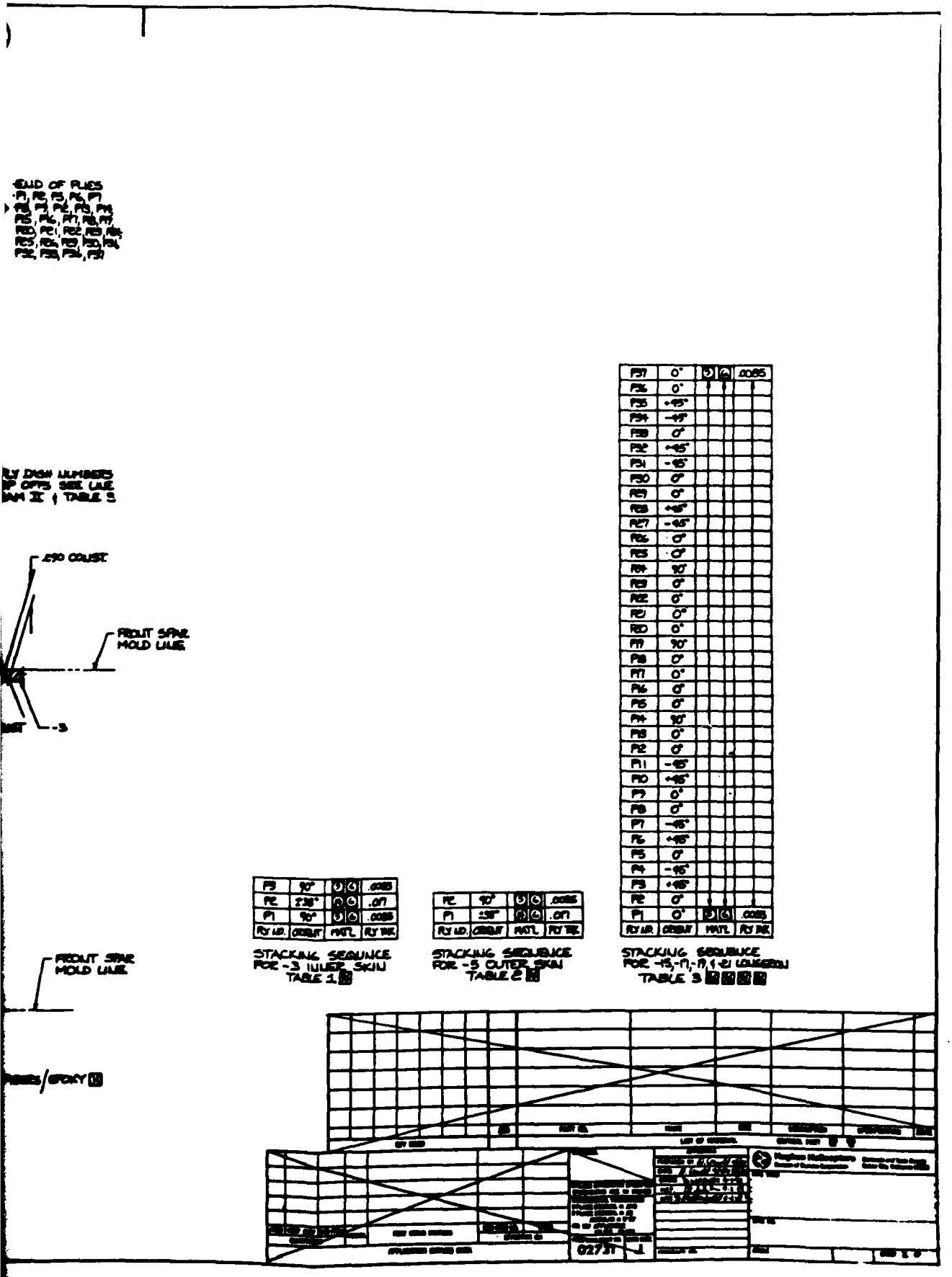
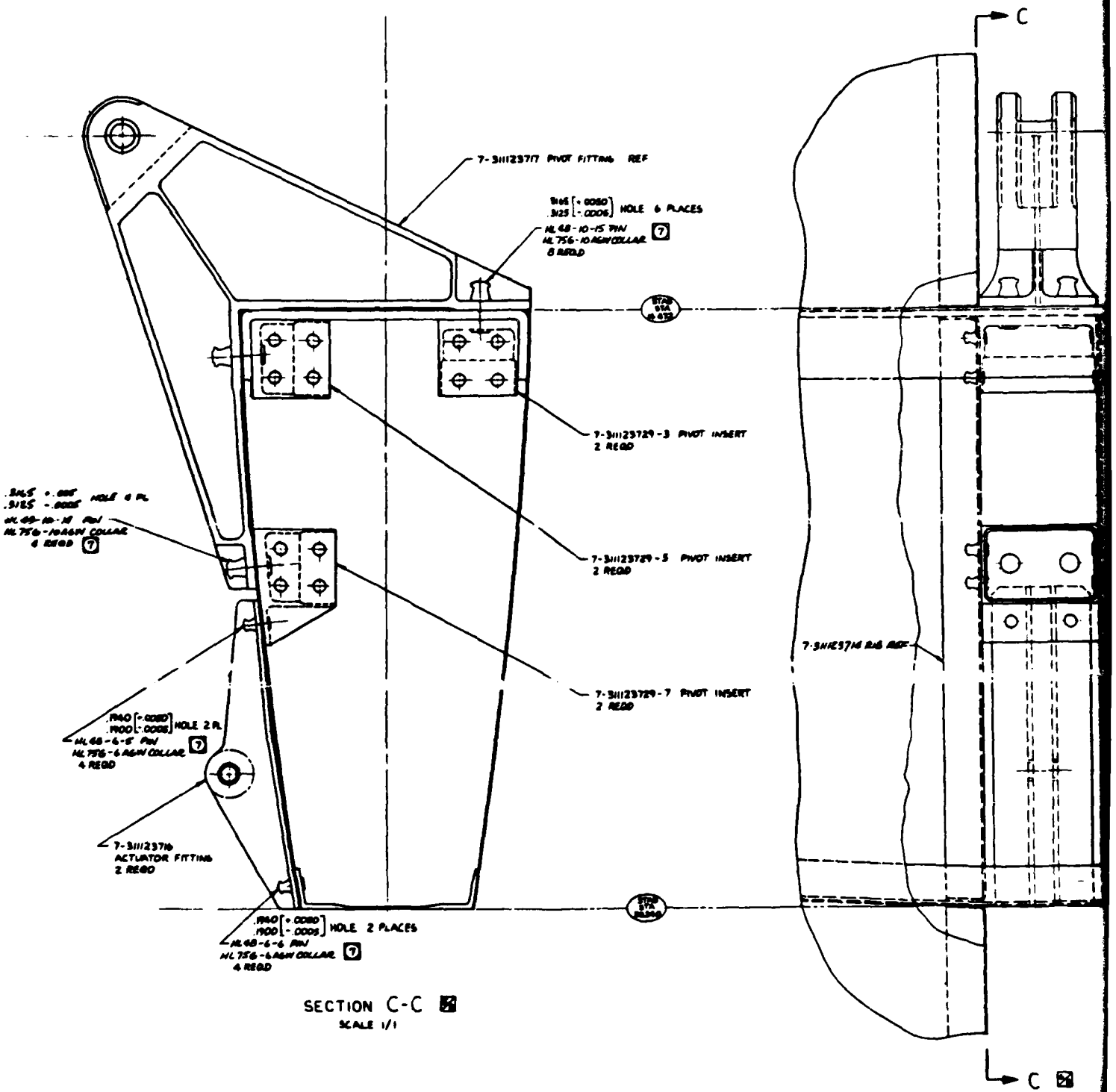
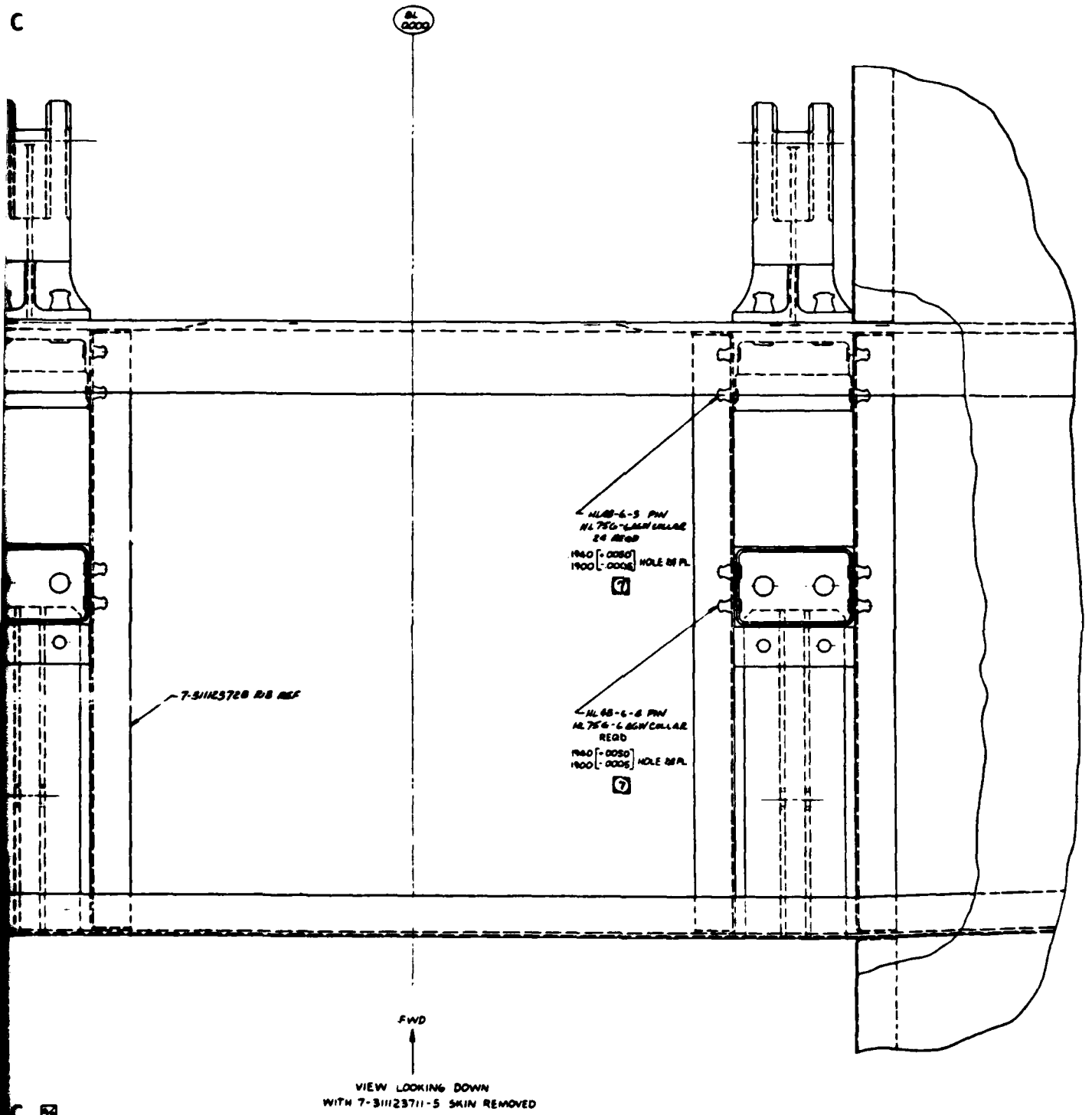


Figure 98. Vertical Tail Subassembly (Sheet 2 of 2)



SECTION C-C

SCALE 1/1


BL 0000
HL88-6-5 PIN
HL756-6 COLLAR
24 REQD
1940 [+.0050 -.0005] HOLE 88 PL
7
7-3111237128 RIB REF
HL88-6-6 PIN
HL756-6 COLLAR
REQD
1960 [+.0050 -.0005] HOLE 88 PL
7
FWD
VIEW LOOKING DOWN
WITH 7-3111237111-5 SKIN REMOVED

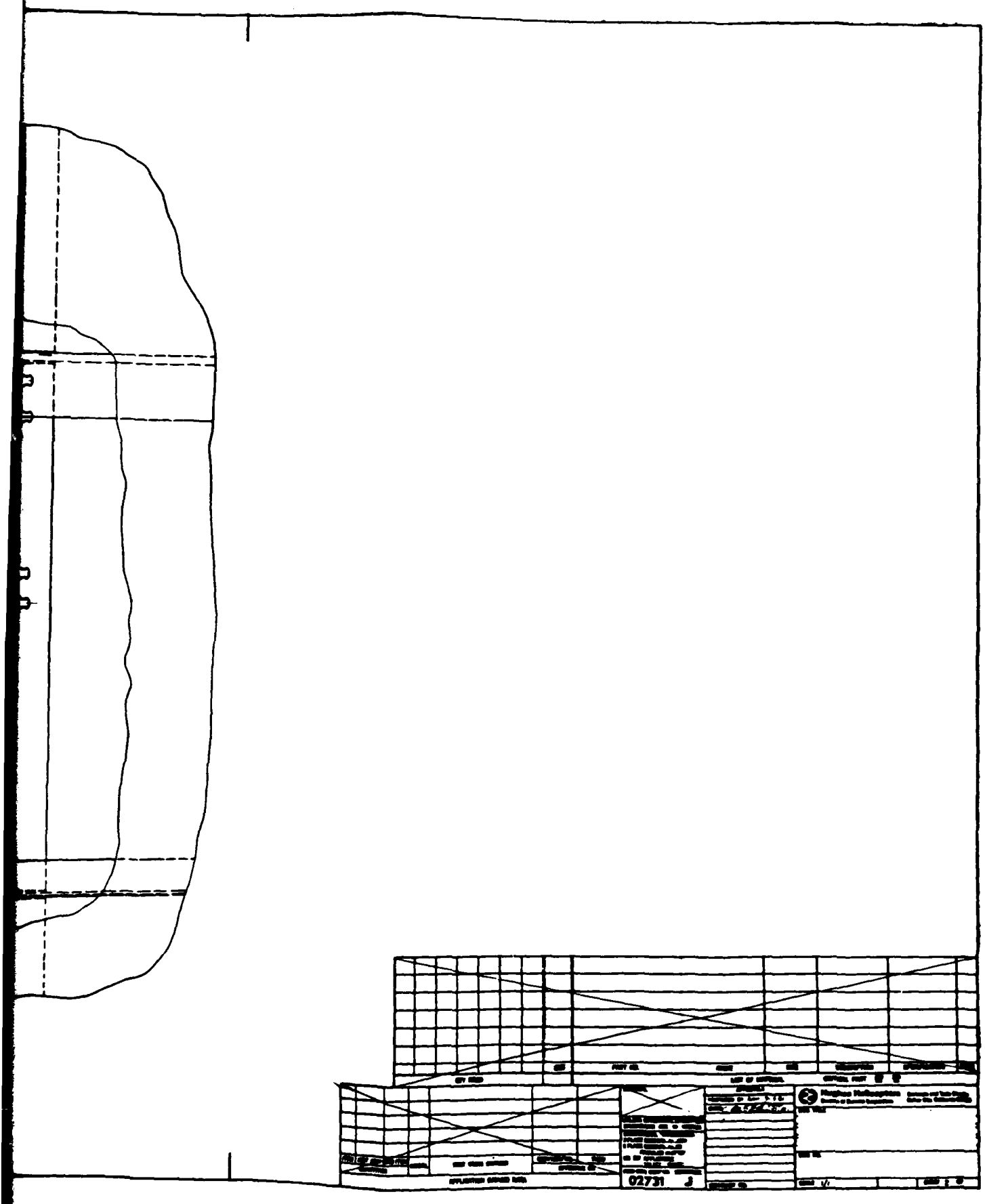
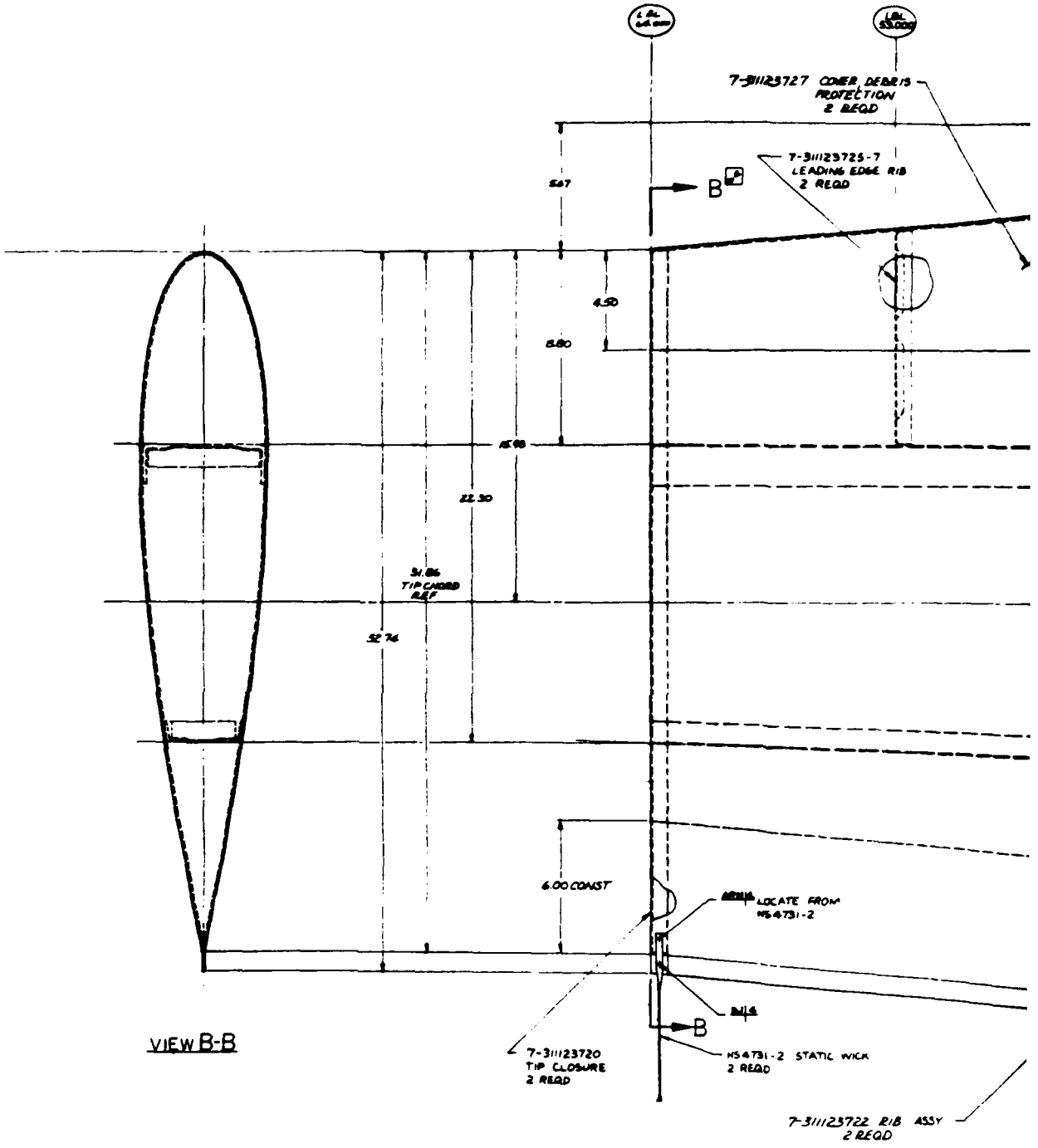
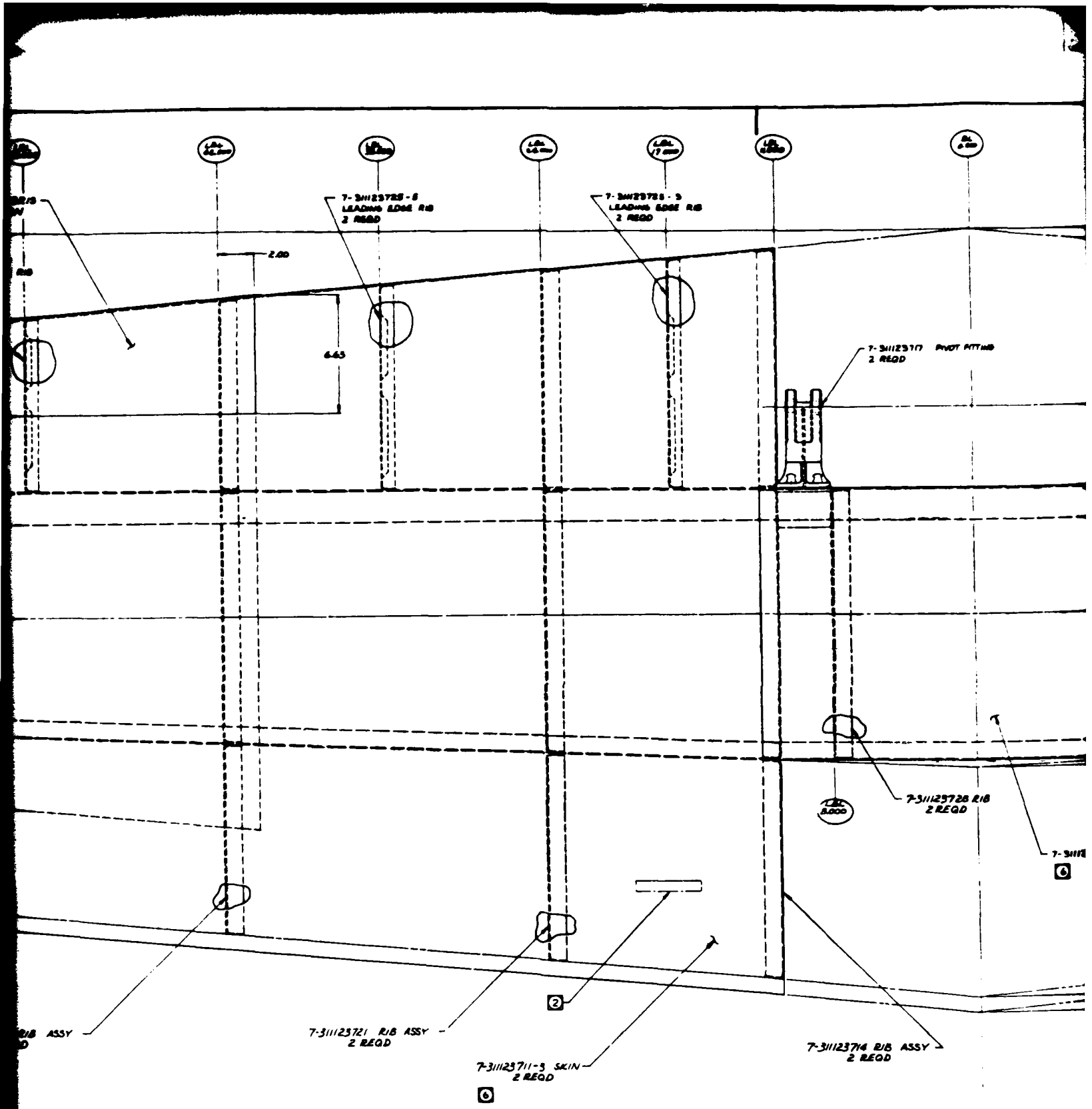
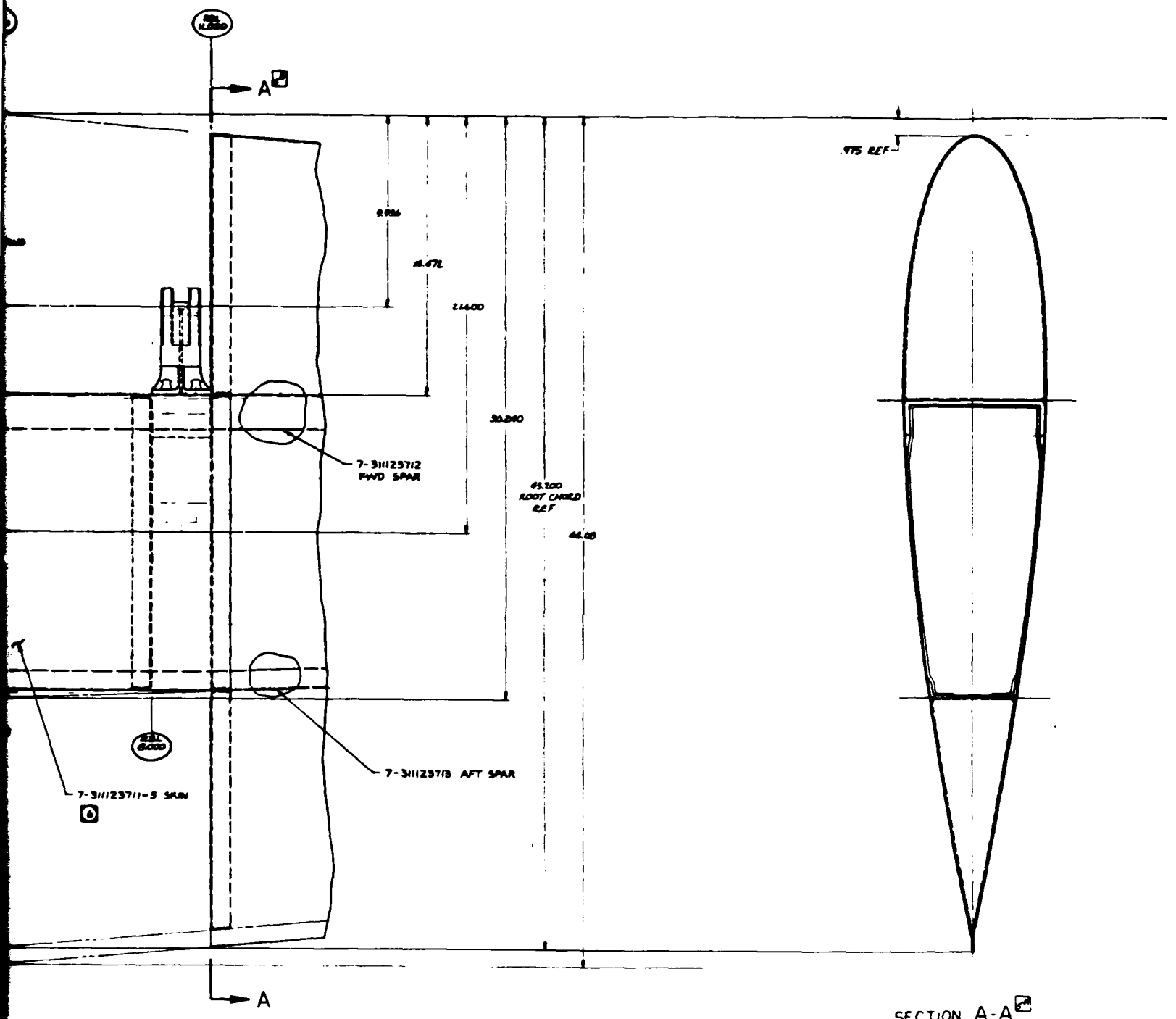


Figure 99. Stabilator Assembly (Sheet 1 of 2)


7-311123727 COVER DEBRIS PROTECTION 2 REQD
7-311123725-7 LEADING EDGE RIB 2 REQD
B
5.47
4.50
8.80
15.98
22.30
31.86 TIP CHORD REF
32.74
6.00 COAST
LOCATE FROM MS4731-2
B
7-311123720 TIP CLOSURE 2 REQD
MS4731-2 STATIC WICK 2 REQD
7-311123722 RIB ASSY 2 REQD

VIEW B-B


7-311123725-5
LEADING EDGE RIB
2 REQD
7-311123725-3
LEADING EDGE RIB
2 REQD
2.00
6.63
7-311123717 PIVOT FITTING
2 REQD
7-311123728 RIB
2 REQD
7-311123721 RIB ASSY
2 REQD
7-311123711-3 SKIN
2 REQD
7-311123714 RIB ASSY
2 REQD
2
6
6


A
9.986
18.672
24.600
30.240
43.200
ROOT CHORD REF
44.08
7-31112S712
FWD SPAR
7-31112S713 AFT SPAR
7-31112S711-5 SKIN
6
A
.975 REF

SECTION A-A

9 FINISH NO ED PER EPB 4-230
8 SEAL FASTENERS USING HMS 16-1097 PER HP 5-10 TYPE IV
7 INSTALL HI-LOKS PER HP 3-2
6 INSTALL BLIND RIVETS PER HP 3-6
5 INSTALL SOLID RIVETS PER HP 3-0
[4] BOND PER HP16-30
3 DIMENSIONS & TOLERANCES PER ANSI Y14.5
[2] IDENTIFY PER HP8-5 TYPE I, CLASS 3 LOCATE APPROX AS SHOWN
1 CONTOUR PER 7-2L1123600 EXCEPT PLANFORM AS SHOWN
NOTES UNLESS OTHERWISE SPECIFIED

[4]			
AR		ADHESIVE	FILM
2	CR 3213	RIVET	
2	MS20470AD6	RIVET	
56	HL 756-6 AGW	COLLAR	
12	HL 756-10AGW	COLLAR	
4	HL 48-6-6	PIN	
4	HL 48-6-5		
24	HL 48-6-4		
24	HL 48-6-3		
8	HL 48-10-15		
4	HL 48-10-14	PIN	
2	MS4731-2	STATIC WICK	
2	7-311123716	FITTING ACTUATOR	
2	7-311123729	INSERT, FITTING	
2	7-311123727	DEBRIS PROTECTION	
2	7-311123728	RIB, BL 8.00	
2	7-311123725	RIBS, LEADING EDGE	
2	7-311123722	RIB ASSY BL 42.00	
2	7-311123721	RIB ASSY, BL 24.00	
2	7-311123720	TIP CLOSEOUT	
2	7-311123717	PIVOT FITTING	
2	7-311123714	RIB ASSY, BL 4[illegible]	
1	7-311123713	SPAR, AFT	
1	7-311123712	SPAR, FWD	
1	7-311123711	SKIN	
DSC	PART NO.	NOMENCLATURE	

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Figure 99. Stabilator Assembly (Sh

D
-1097 PER HP 5-10 TYPE IV

-E
-D

PER ANSI Y14.5
CLASS 3 LOCATE APPROX AS SHOWN
CEPT PLANFORM AS SHOWN
IFIED

AR			ADHESIVE		FILM	HMS 16 1111 CLASS 2	
2		CR 3213	RIVET				
2		MS20470AD4	RIVET				
56		HL 756-6 AGW	COLLAR				
12		HL 756-10AGW	COLLAR				
4		HL 48-6-6	PIN				
4		HL 48-6-5					
24		HL 48-6-4					
24		HL 48-6-3					
8		HL 48-10-15					
4		HL 48-10-14	PIN				
2		HS4731-2	STATIC WICK				9C
2		7-311123716	FITTING ACTUATOR				17C
2		7-311123729	INSERT, FITTING				16D 16E
2		7-311123727	DEBRIS PROTECTION				8G
2		7-311123728	RIB, BL 8.00				6D
2		7-311123725	RIBS, LEADING EDGE				[illegible]
2		7-311123722	RIB ASSY BL 42.00				8B
2		7-311123721	RIB ASSY, BL 24.00				7B
2		7-311123720	TIP CLOSEOUT				9C
2		7-311123717	PIVOT FITTING				6G
2		7-311123714	RIB ASSY, BL 11.6				6B
1		7-311123713	SPAR, AFT				5D
1		7-311123712	SPAR, FWD				5F
1		7-311123711	SKIN				7B
BSC	[illegible]	PART NO.	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]

LIST OF MATERIAL

Hughes Helicopters

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SHEET 1 OF 2

Figure 99. Stabilator Assembly (Sheet 2 of 2)